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XV-9A HOT CYCLE RESEARCH AIRCRAFT PROGRAM

SUMMARY REPORT

By

S. Cohan

N. B. Hirsh

June 1966

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-877(T)

HUGHES TOOL COMPANY

CULVER CITY, CALIFORNIA

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This report has been prepared by Hughes Tool Company, Aircraft Division, under the provisions of Contract DA 44-177-AMC-877(T). It presents in summary form the results of the XV-9A hot cycle research aircraft program. The report is published for the dissemination of information and the reporting of program results.

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ABSTRACT

This report summarizes a research program covering the design, fabrication, and test of the XV-9A Hot Cycle Research Aircraft.

Discussion of the program is broken into five major areas: design and fabrication, engine and whirl tests, component testing, ground tests, and flight tests. During the program, conducted from 29 September 1962 through 15 March 1965, the flight feasibility of the Hot Cycle rotor was successfully validated.

FOREWORD

This report was prepared in accordance with Contract DA 44-177-AMC-877(T) with the U. S. Army Aviation Materiel Laboratories. The contract became effective on 29 September 1962. Work was completed on 15 March 1965.

The work was accomplished by the Hughes Tool Company-Aircraft Division in Culver City, California, under the direction of Mr. H. O. Nay, Director of Engineering, and under the direct supervision of Mr. C. R. Smith, Manager, Hot Cycle Department. This report was prepared by S. Cohan and N. B. Hirsh.

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SUMMARY

The design, fabrication, and test of the XV-9A Hot Cycle Research Aircraft (Figure 1) have been accomplished in accordance with U.S. Army Aviation Materiel Laboratories Contract DA 44-177-AMC-877(T). This report summarizes the program and delineates significant results, conclusions, and recommendations derived therefrom.

The highly successful XV-9A program has established the feasibility, practicality, and attractiveness of the Hot Cycle propulsion system and has provided valuable data for use in design of operational aircraft using the Hot Cycle system. All program objectives were achieved, and the program was completed with no major incidents or delay. The flight and operational characteristics of the aircraft and propulsion system were generally as predicted. No changes to the rotor system were required, and only minor system adjustments and modifications were necessary.

Engine and rotor whirl tests were performed in two phases from October 1963 through May 1964. Engine testing demonstrated satisfactory twin-engine operation with mixed gas flow through a common exhaust duct, isolation of individual engines, and engine operation at various steady and transient conditions. Whirl tests demonstrated rotor operation through the full range of cyclic and collective pitch travel and demonstrated a measured rotor lift of 23,000 pounds with the ground-test gas generators. The performance of all systems and equipment during testing was satisfactory.

A total of 70 hours 9 minutes of testing was accomplished. This figure represents 17 hours 30 minutes of individual engine operation with diverter valves overboard, 7 hours 48 minutes of twin-engine operation with mixed gas flow into a common exhaust duct and exit nozzle, 15 hours 49 minutes of rotor operation on the whirl tower, 13 hours 20 minutes of rotor operation in the aircraft ground test, and 15 hours 42 minutes of flight testing. The test programs were successfully completed with no major incidents, delays, failures, or operational problems associated with the aircraft, propulsion system, or the associated components.

Preflight and tie-down tests were conducted from 10 August 1964 through 4 November 1964, with satisfactory results.

The component tests conducted during the program demonstrated the functional and structural adequacy of selected rotor and propulsion system components for the planned flight test program.

The flight test program was conducted from 5 November 1964 through 5 February 1965. It consisted of 21 flights for a total flight time of 15 hours 42 minutes.

The discussion of the XV-9A program is broken into five major areas: design and fabrication, engine and whirl tests, component testing, ground tests, and flight tests. Reports have been published for each of these categories, and should be referred to if more detailed information is desired (References 1 through 4).

Figure 1. XV-9A Hot Cycle Research Aircraft.

RESULTS AND CONCLUSIONS

The XV-9A Hot Cycle Research Aircraft flight test program definitely substantiated the feasibility of the Hot Cycle system for integration into an operational-type aircraft having a high payload to empty weight capability.

Flying and control qualities were adequate for the research application of the XV-9A aircraft. Moderate hub restraint would provide a substantial improvement in the flying qualities of the aircraft.

In general, flight test results substantiated the XV-9A helicopter design loads. The vibration level was reduced by the redistribution of the ballast in the fuselage.

The overall external sound level was closely comparable with that of a turboshaft helicopter of similar size. Sound power level measured in the cockpit during flight was not objectionable to the pilot, and radio communications were satisfactory.

The hover performance with variable tip-cascade nozzles agrees very well with whirl test data using fixed-area nozzles and with previously predicted performance of the rotor.

Fuel consumption measurements agree with predicted rotor lift values below approximately 12, 000 pounds, but are slightly higher than predicted at higher values of rotor lift.

None of the temperatures of the structure, ducting, or gas-generator components exceeded the values for which allowances had been made in design.

In general, the airplane systems performance was very satisfactory. Some instances of minor electrical and component difficulties were encountered, as might be expected in a research and development program. No failures of any consequence occurred in flight.

The XV-9A yaw-valve system provided adequate low-speed directional control and was mechanically satisfactory, but resulted in high power losses during operation in an open position. Because of the high power losses, it may be concluded that a pneumatic yaw-control valve system would not be suitable for use in an operational Hot Cycle aircraft.

As a result of the apparent limited service life of the rotor blade spars it may be concluded that further testing is required prior to use of this type of Hot Cycle blade design in an operational aircraft.

RECOMMENDATIONS

Three major recommendations are detailed in the following paragraphs.

1. Exploit in an operational aircraft the major improvements in helicopter simplicity, light weight, payload capacity, and economy provided by the Hot Cycle rotor propulsion system. As a major part of such a program, the following should be accomplished.

   a. Survey various rotor structural configurations to accommodate the higher temperatures and pressures of the advanced-type gas generators. Include the use of a restrained tilting hub or a hub with offset hinges to provide increased control power and damping.

   b. Size the diverter valves to match the powerplant, and design for minimal leakage to negate power losses.

   c. Use a tail rotor or fan in lieu of the yaw valve employed on the XV-9A. This would provide for improved yaw control and for a reduced power requirement.

   d. Conduct a study of the total power management problem to coordinate gas-generator power during transient as well as during steady-state operation. This represents a most important field for future Hot Cycle applications.

   e. Conduct a structural development test program in order to accumulate design criteria necessary for development of advanced Hot Cycle rotor systems having greatly increased service life of rotor components subject to vibratory loads. Particular emphasis would be given to the blade spars and their attachment configuration.

   f. Design and construct a hot-gas facility to be used in the development of optimum Hot Cycle propulsion system components such as blade-tip cascades, hot-gas seals, blade segments, hot-gas valves, bellows, and so on.

   g. Develop and test various improved cascade designs in order to obtain the optimum configuration for use on future Hot Cycle rotor systems.

h. Conduct a program to develop improved hot-gas seals in order to obtain seals with greatly increased service life and maintainability.

i. Conduct an investigation to determine the rotor dynamics and flying qualities of the proposed heavy-lift rotor system design. The program would consist of an integrated program of analysis, model design, construction, and test, with appropriate modification being made to the model to establish a rotor configuration with optimum dynamic characteristics.

2. In the 20-hour follow-on flight test program:

   a. Change the directional control system to permit full range of yaw-control valve opening.

   b. Install an engine low-speed warning system.

   c. Install generator cooling system. Incorporate ram air inlet for forward flight.

   d. Add increased cooling capabilities to both the fuselage pylon and cowling to compensate for higher ambient temperature operation.

3. Carry out additional testing of the XV-9A at the conclusion of the 20-hour follow-on flight test program, as follows:

   a. Conduct rotor-system tether test to measure component performance more accurately by supplying gas to all three blades, rather than to only one as was done in the previous tether test.

   b. Conduct a cleanup (refairing) on the leading edge and tip of the XV-9A rotor blades. Evaluate the effect on profile power by short hover flights prior to and subsequent to the modification.

   c. Conduct a teardown inspection of the XV-9A rotor system and aircraft to determine the condition of various components as to structural distress due to heat and fatigue loadings, wear of seals, condition of bearings, linkages, and so on, for application to future Hot Cycle rotor system designs.

XV-9A RESEARCH AIRCRAFT DESIGN AND FABRICATION

DISCUSSION

The XV-9A Hot Cycle Research Aircraft (Hughes Tool Company Model 385) utilizes the 55-foot-diameter Hot Cycle pressure jet rotor system designed, fabricated, and tested under Contract AF 33(600)-30271. The aircraft was designed in general accordance with References 5 through 7 for the mission of demonstrating the flight feasibility of the Hot Cycle propulsion system. Maximum use was made of reliable design techniques and off-the-shelf components. The general arrangement of the aircraft is shown in Figure 2. The aircraft design gross weight is 15,300 pounds. The aircraft design overload gross weight (with external payload) is 25,500 pounds; however, the aircraft did not incorporate mechanical provisions for lifting an external load. A complete listing of aircraft characteristics and design criteria is presented on the following pages.

The Hot Cycle propulsion system, shown in Figure 3, operates on the pressure jet principle and provides the simplest possible helicopter rotor-drive system. Two YT-64 gas generators provide the high-energy gases for rotor power. The gas-generator exhaust gases are ducted through diverter valves, stationary ducts, a trifurcated rotating duct in the rotor hub, and the blade ducts to the blade-tip-cascade nozzles. The rotor-control system is a full-power system, with three hydraulic servo cylinders providing actuating forces for collective and cyclic inputs to the rotor. As a result of the absence of significant rotor drive-shaft torque, no tail rotor is required. A jet-reaction yaw-control valve, mounted in the aft fuselage, is powered by the hot gases from the gas generators and supplies the required yaw-control force during hover and low-speed forward flight. Aerodynamic rudder-control surfaces are used for yaw control at higher forward flight speeds.

AIRCRAFT CHARACTERISTICS

Overall Dimensions

Rotor diameter	55.00 feet
Aircraft length (rotor turning)	59.70 feet
Fuselage length	44.17 feet
Tread of main wheels	11.00 feet
Height (to top of rotor hub)	12.40 feet
Width (across lateral pylons)	12.20 feet

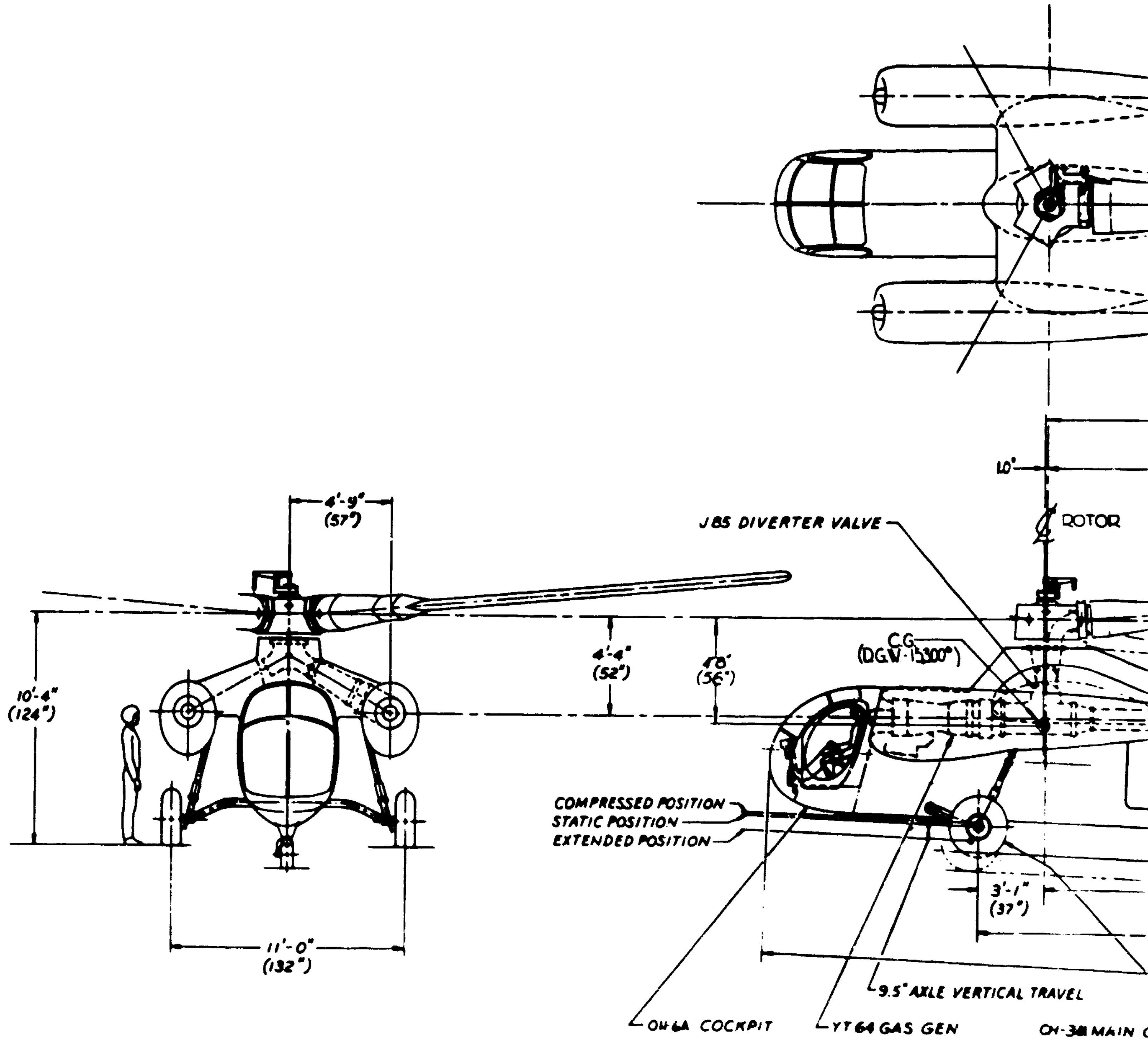
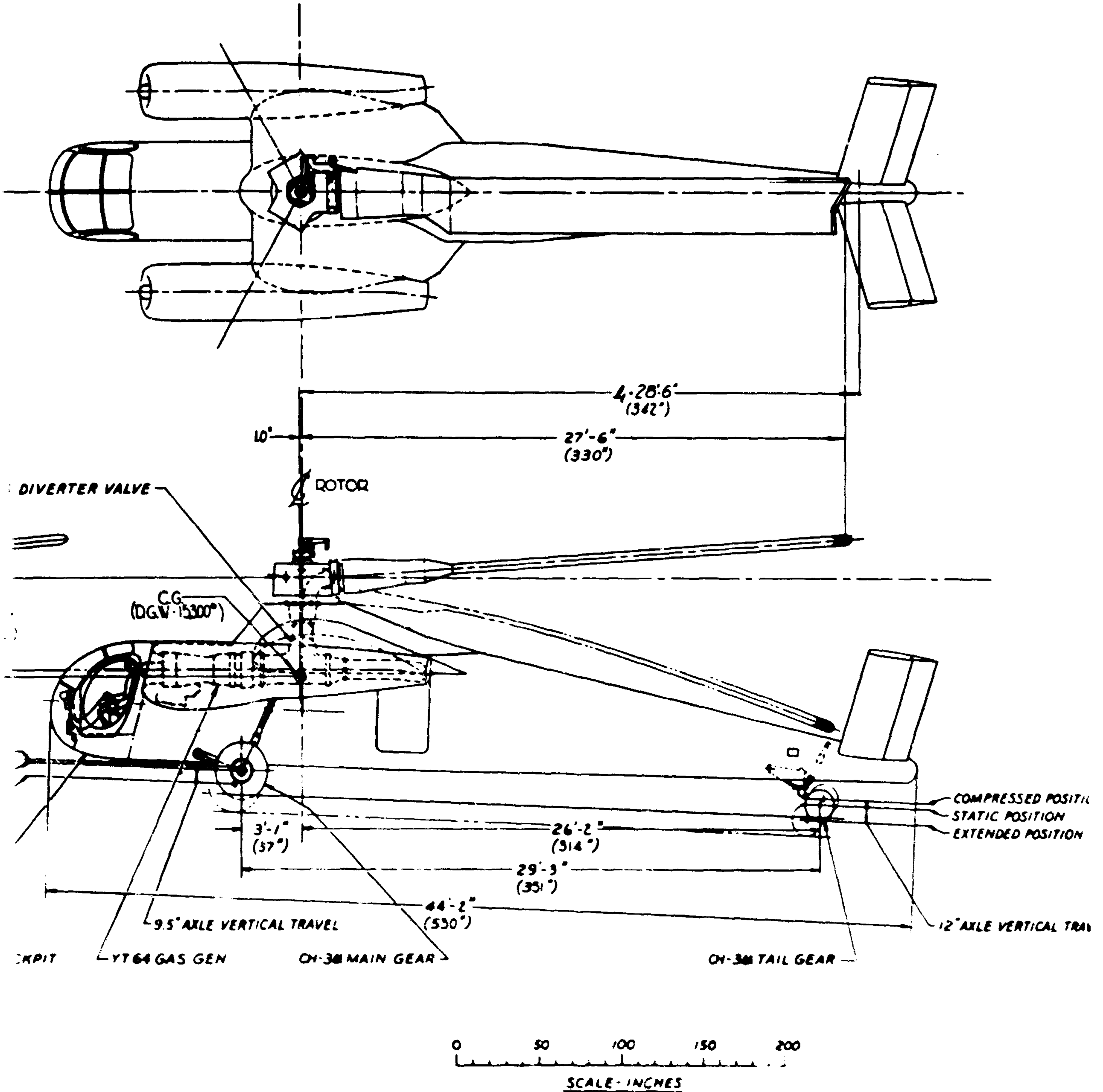


Figure 2. General Arrangement - XV-9A Hot Cycle Research Aircraft.



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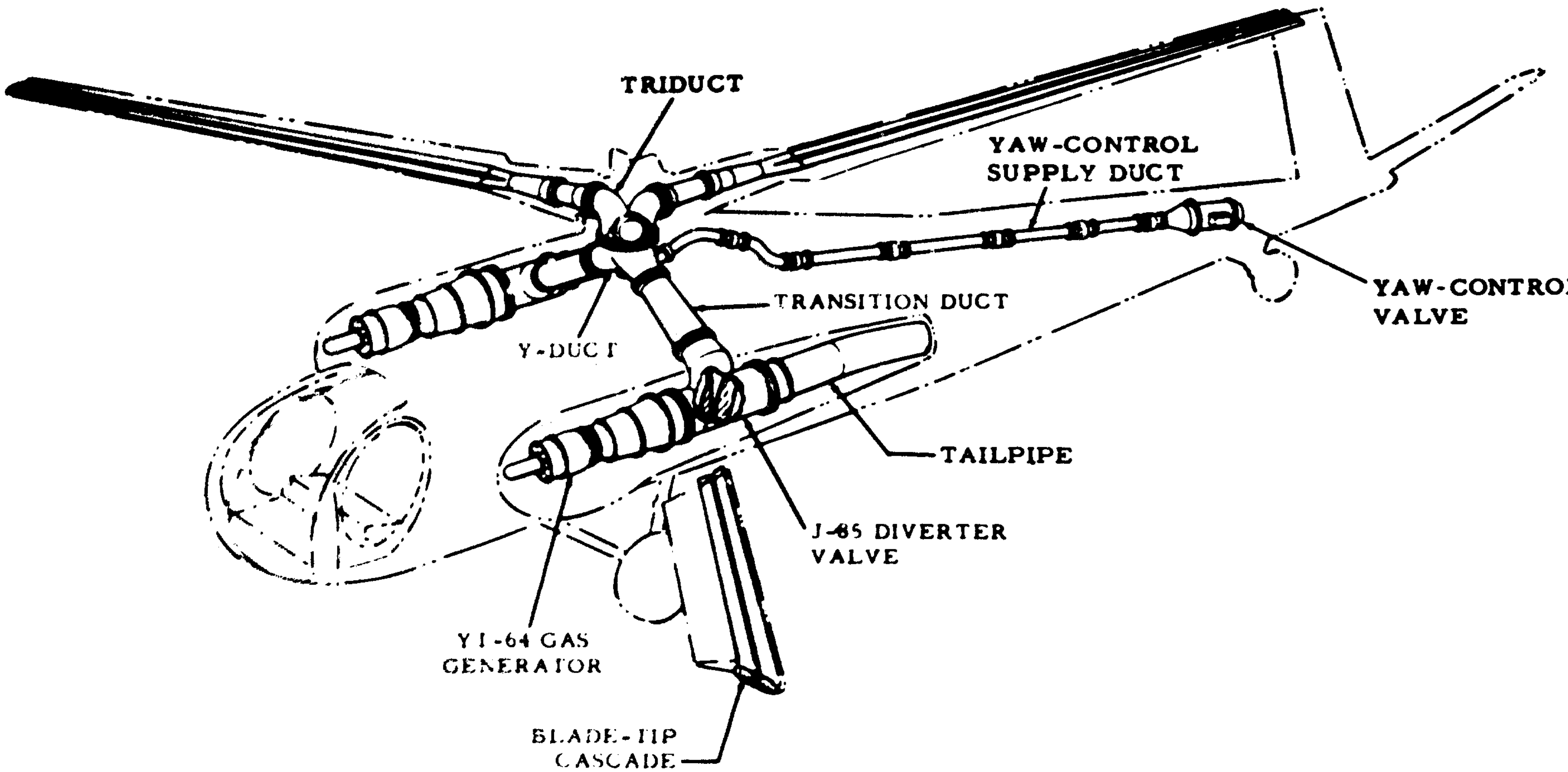


Figure 3. General Arrangement - XV-9A Propulsion System.

Weight Summary

Empty weight	8,641 pounds
Design minimum gross weight	10,000 pounds
Design gross weight (internal cargo)	15,300 pounds
Design alternate overload gross weight (external cargo)	25,500 pounds

Design Performance

Condition	Gross Weight	Altitude and Temperature	Speed
Design maximum speed	15,300 pounds	SL Standard	140 knots
Design maximum speed	10,000 pounds	SL Standard	150 knots
Design maximum dive speed	15,300 pounds	SL Standard	200 knots
Design maximum dive speed	10,000 pounds	SL Standard	200 knots

Design Loads and Load Factors

Maneuver	2.5 g limit at design gross weight
Ground flapping	
Blade droop stop and hub 9-degree tilt stop	2.5 g limit
Hub 2-degree tilt stop	2.0 g limit

Rotor

Type	Floating hub, coning blade
Number of blades	3
Rotor diameter	55 feet
Blade area (total three blades)	217.5 square feet
Disc area	2,392 square feet
Rotor solidity	0.091
Blade chord	31.5 inches
Blade airfoil	NACA 0018
Blade twist	-8 degrees

Hot-gas ducts

Number of ducts per blade	2
Total duct area in each blade	54.8 square inches
Duct utilization, $\frac{\text{duct area}}{\text{blade-cross-section area}}$	0.451
Tip-nozzle area per blade (closure valve open)	37.5 square inches

Rotor speeds

	rpm	V_{tip} (fps)
Design operational, power-on or power-off	243	700
Design minimum, power-on	225	648
Design maximum, power-on (red line)	255	734
Design minimum, power-off	225	648
Design maximum, power-off (red line) (1.1 x maximum power-on rpm)	280	807
Rotor-speed limit, power-on or power-off (1 1 x maximum power-on rpm) x 1.05	295	848

Powerplant

			Two YT-64-6 Gas Generators		
	T (°R)	T (°F)	Pressure Ratio	Pressure (psig)	Gas Flow (lb/sec)
SLS normal rated	1,499	1,039	2.61	23.6	23.8
SLS military rated	1,577	1,117	2.83	26.9	25.0

Empennage

Area (true)(total)	54.00 square feet
Dihedral	45.00 degrees
Sweep	7.50 degrees
Incidence (with respect to rotor shaft)	1.00 degree ±5.00-degree adjustment
Chord	3.50 feet
Span (true)	15.40 feet
Aspect ratio (geometric)	4.35
Airfoil	NACA 0012

Rudder chord (37.5 percent including overhang)	1.31 feet
Rudder span (true)	15.40 feet
Rudder area (true)	19.90 feet

Maximum Control Displacement

Cyclic control	
Longitudinal cyclic pitch travel	±10 degrees
Longitudinal cyclic stick travel	Total, 13 inches
Lateral cyclic pitch travel	±7 degrees
Lateral cyclic stick travel	Total, 12 inches
Collective control	
Collective pitch travel (at 75-percent rotor radius)	-2 degrees to +10 degrees
Collective stick travel	7.5 inches
Rudder pedal (from neutral)	
Full left	3.0 inches
Full right	3.0 inches
Rudder deflection	±20 degrees
Yaw valve area	24 square inches

ROTOR SYSTEM

The Hot Cycle rotor, shown schematically in Figure 4 and complete in Figure 5, consists of three coning blades retained by straps to the hub, which in turn is attached to the rotor shaft through a gimbal. The shaft is mounted to the aircraft structure through two sets of bearings, displaced in the vertical direction, which react moment, side load, and rotor thrust.

The constant section of the blade, shown in Figure 6, consists of two air-cooled laminated steel spars separated chordwise by and bolted to eighteen identical sheet metal duct segments. Segments are joined by flexible couplings, with the skin left slip-jointed. In this structural arrangement, only the spars react blade bending moments, centrifugal loads, and unbalanced spanwise duct pressure loads. Torsional and chordwise shear loads are carried by the assembly of segments. Steel leading edge sections and aluminum trailing edge sections complete the blade.

The blade-root section is a continuation of the structural arrangement described above, with a feathering ball at the root end to transfer shear loads from the blade to hub through a Fabroid bearing.

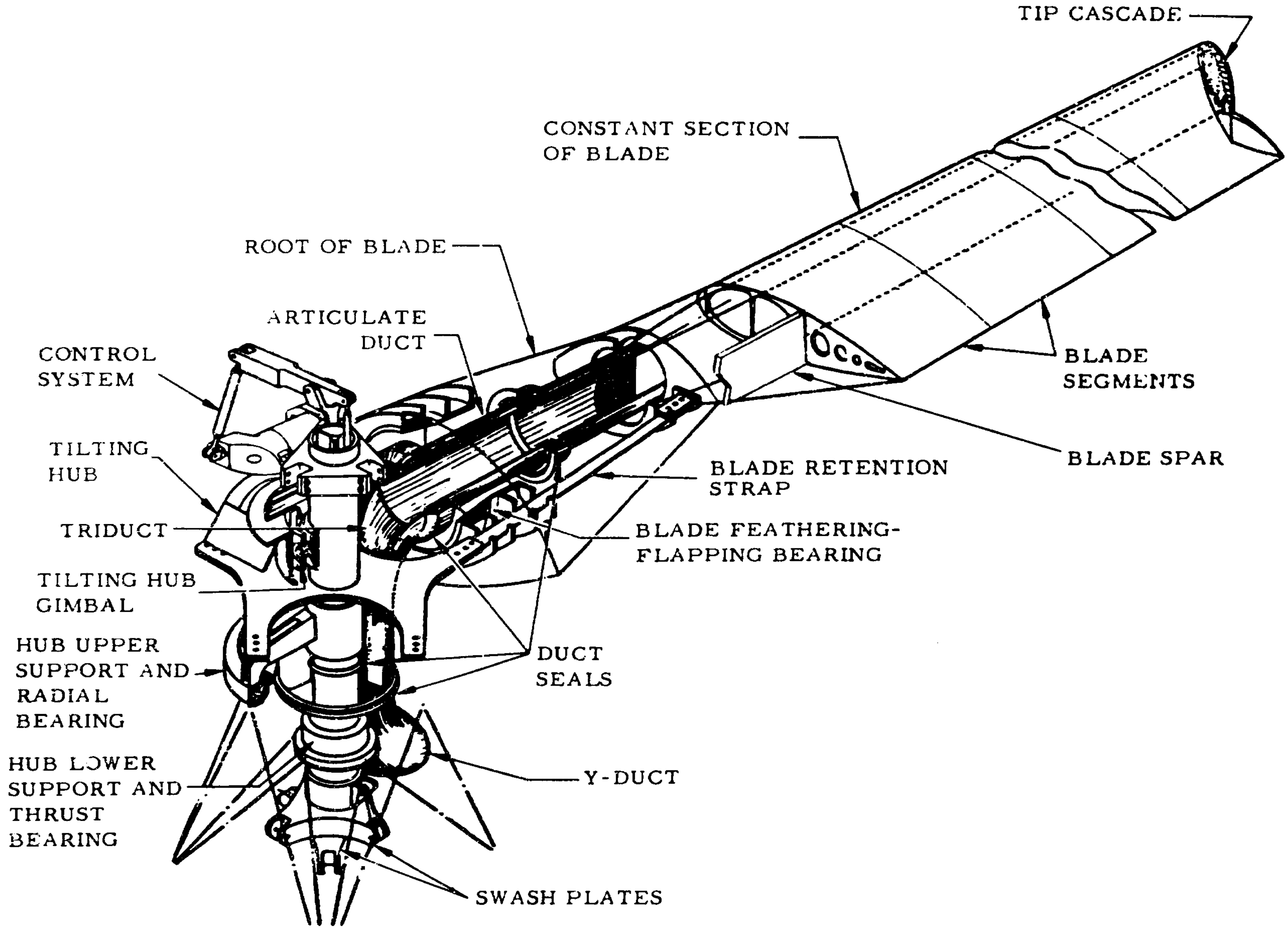


Figure 4. Hot Cycle Rotor System Schematic.

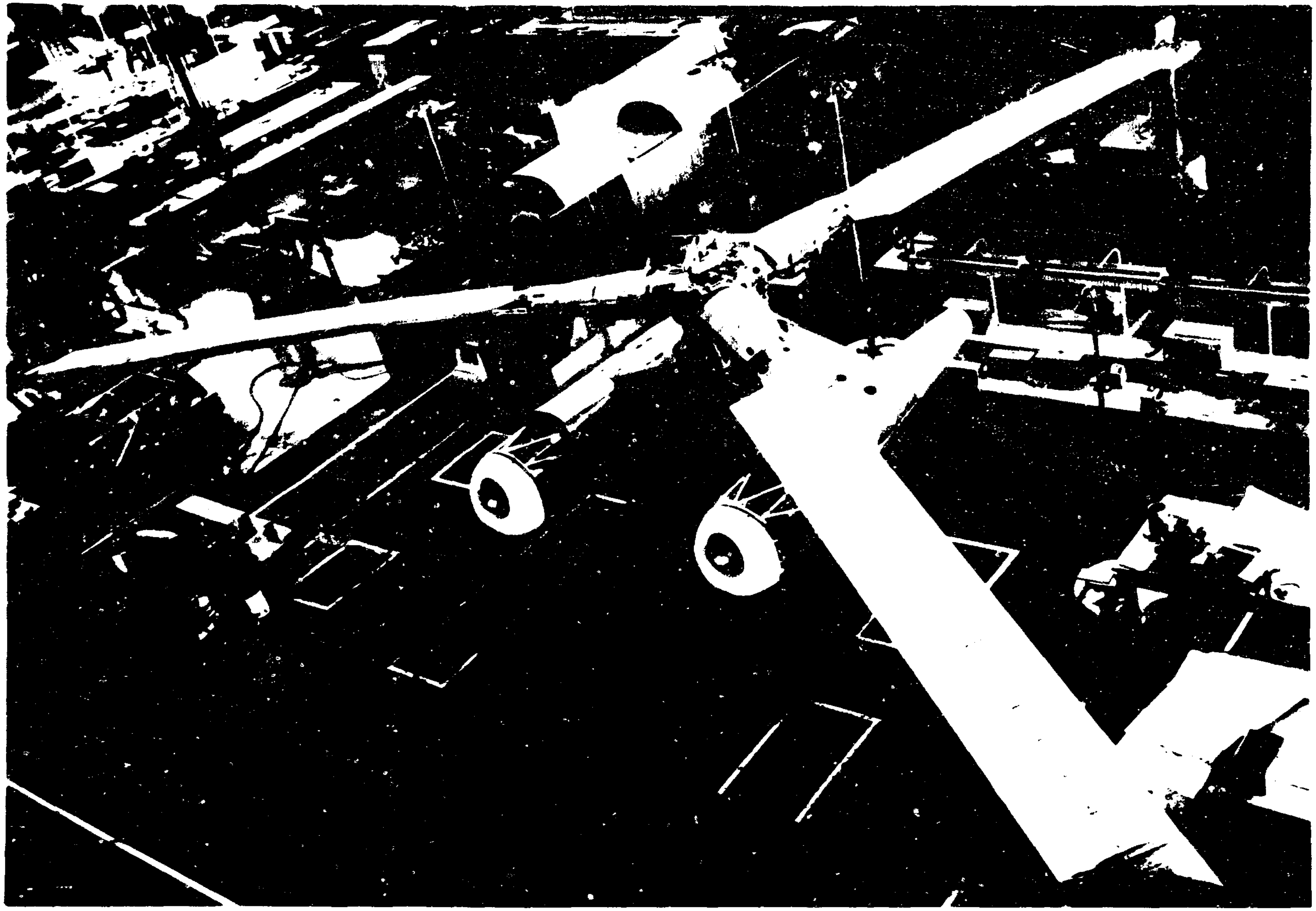


Figure 5. XV-9A Power Module and Rotor.

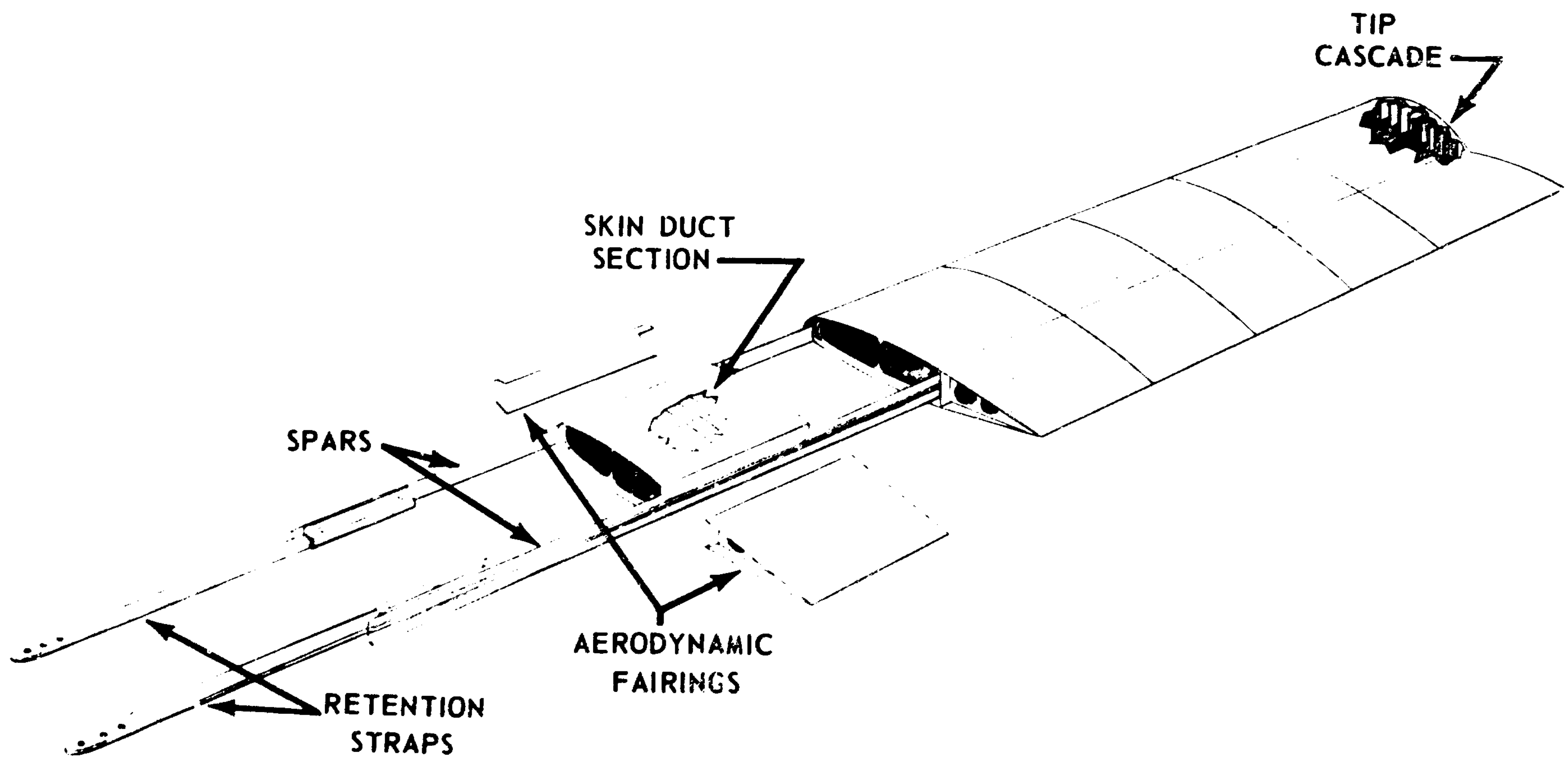


Figure 6. Hot Cycle Rotor Blade.

The hub and shaft form the central pivot for the rotor and provide support for the rotating portion of the control system. Each blade is attached to the hub by a pair of tension strap packs, which restrain the blade in the chordwise and centrifugal load direction but allow freedom in coning and feathering. The gimbal-mounted free-floating hub ties the blades together and transfers the resultant load to the rotating shaft, which is supported by an upper radial bearing and by a lower thrust bearing. A circulating oil system is used to lubricate and cool the bearings.

The rotor duct system is shown schematically in Figure 7. The stationary Y-duct is supported from the radial bearing support, while the rotating triduct is supported from the rotor shaft. Carbon is used as the sealing material for these rotating ducts, and for the blade articulating duct seal, which allows blade flapping motion. Feathering motion and expansion are provided for by the outboard seal on the articulating duct. This seal consists of a nest of slotted René 41 lip laminations riding on a tungsten-carbide-coated cylinder. From the articulating duct, the hot gas flows through a transition duct into the blade ducting and to the blade tip cascade.

The tip-cascade assembly turns the gas flow to drive the rotor. The tip-cascade area can be closed to approximately one-half the exit area for single-engine operation. This adjustment is accomplished pneumatically by pilot command.

An accessory gearbox, driven by the rotor shaft, drives the rotor lube pump, the rotor tachometer generator, the rotor-speed-governing-system hydraulic pumps, and a control-system hydraulic pump.

PROPULSION SYSTEM

The propulsion system, previously shown in Figure 3, consists of two YT-64-6 turboshaft engines modified to gas generators for application to the XV-9A, the diverter valves, engine controls, and the hot-gas transfer system. The gas generator, the diverter valve, and the exhaust tailpipe are independently mounted, as shown in Figure 8. An interconnecting sliding seal is provided between each component. Each component of the propulsion system can be easily installed or removed without affecting the other components of the system.

The gas-generator assembly includes the basic powerplant, the air inlet, the lubricating system, the starting system, the hydraulic pump, the d-c and tachometer generators, the fuel inlet, the gas-generator

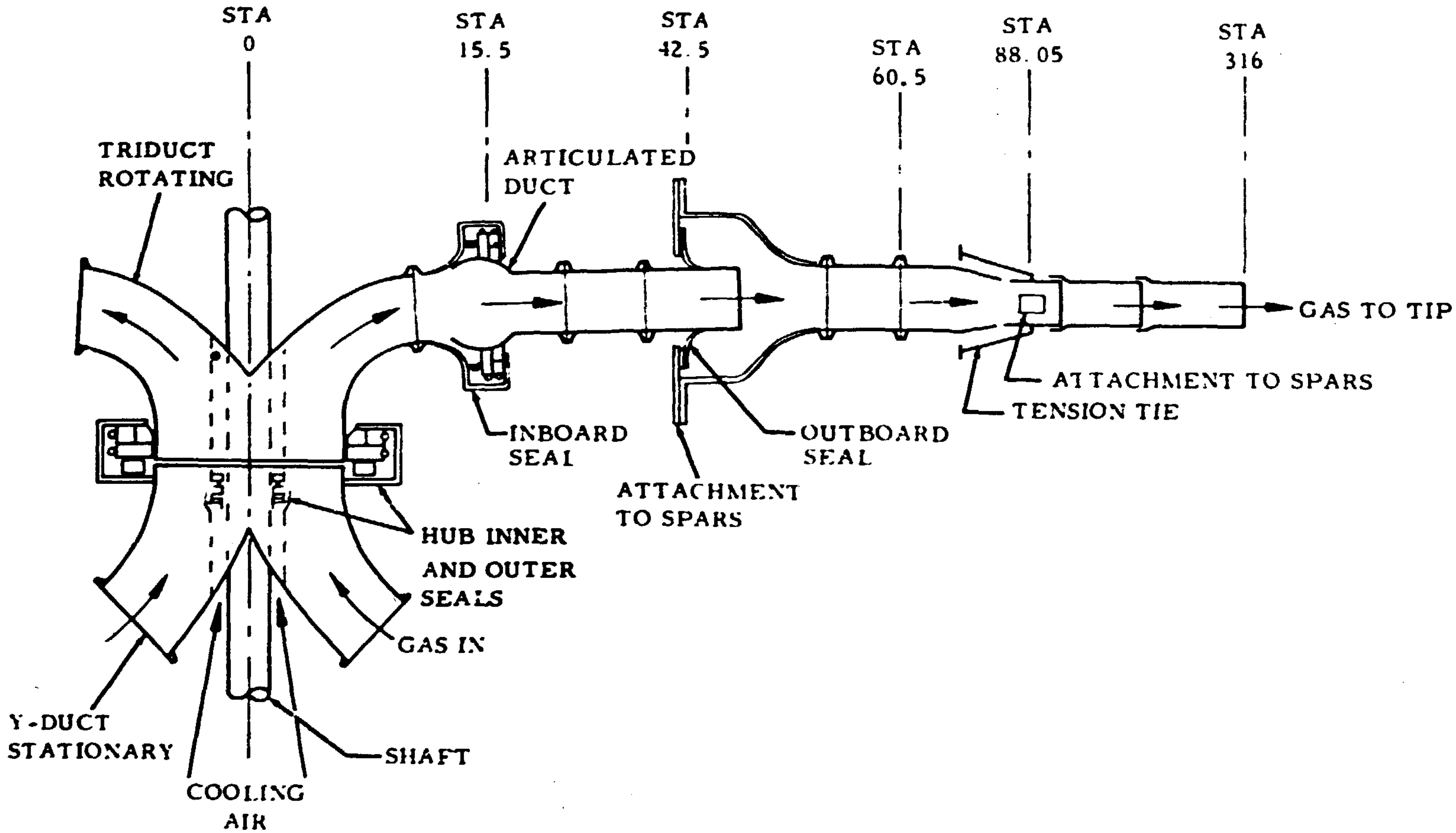


Figure 7. Rotor Hub and Blade Ducting Schematic.

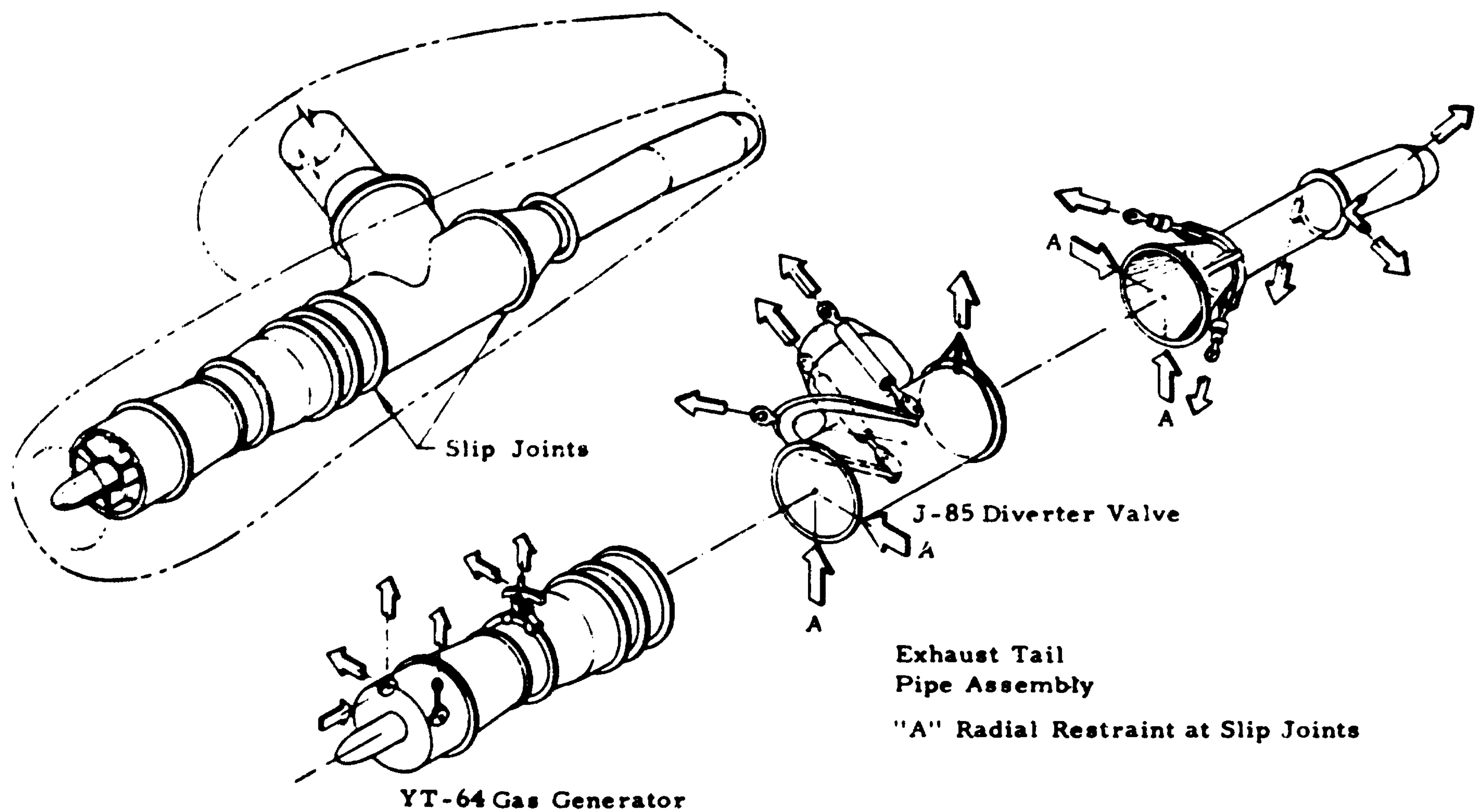


Figure 8. Propulsion System Mounting.

power control attachments, the gas-generator mounts, the fire extinguishing manifold, the fire detection cabling, and the required gas-generator vents and drains. All gas-generator-mounted items, such as accessories, ignition generator, ignitor, filters, and so on, can be inspected, cleaned, adjusted, removed, and/or replaced without the use of special tools or the removal of the engine or primary structure. The gas generator utilizes an integral air impingement starter. A mobile air compressor supplies the required air. Connection is made through an access door in the nacelle cowling. The engine lubrication system includes an oil reservoir integral with the gas-generator inlet. Cooling of the engine oil is accomplished by means of an air-to-oil heat exchanger. Airflow through the heat exchanger is induced by means of an ejector using gas-generator compressor bleed air.

J-85 engine diverter valves are installed in each nacelle immediately aft of the gas generator. The valves are hydraulically actuated by pilot command. Flow may be straight through the tailpipes for engine starting or diverted for rotor operation.

The hot-gas transfer system carries gases from each diverter valve to the rotor duct system and to the yaw-control valve. Ducts are fabricated from corrosion-resistant steel and are insulated. All ducts incorporate bellows to compensate for thermal expansion.

The gas-generator power control system consists of a mechanical power linkage and a rotor-speed-governing feedback link to the engine fuel control. The mechanical power linkage is a manually operated arrangement from the cockpit controls to each engine. Changes to engine setting may be made by movement of the quadrant lever, collective stick, or collective stick twist grip.

Precautionary measures were incorporated in the design of this system, including isolation of combustibles from ignition sources, use of fireproof and fire-resistant lines as well as high-temperature electrical wiring in the engine bays, insulation of all high-temperature gas-carrying ducts, ventilation to induce circulation, and location of drains to negate impingement or reentry. To prevent the spread of fire, the engine bays are isolated from the rest of the airplane, and multiple fuel and hydraulic shutoffs outside the potential fire zone are included.

FUSELAGE STRUCTURE (See Figure 9)

The OH-6A (Hughes Model 369) cockpit structure is used on the XV-9A. It is of conventional sheet metal construction, with emphasis on lightweight structure, visibility, and convenience for the pilot and copilot, seated side by side. Some structural revisions were made to accommodate the XV-9A electrical system, instrumentation, and rotor and propulsion controls installations.

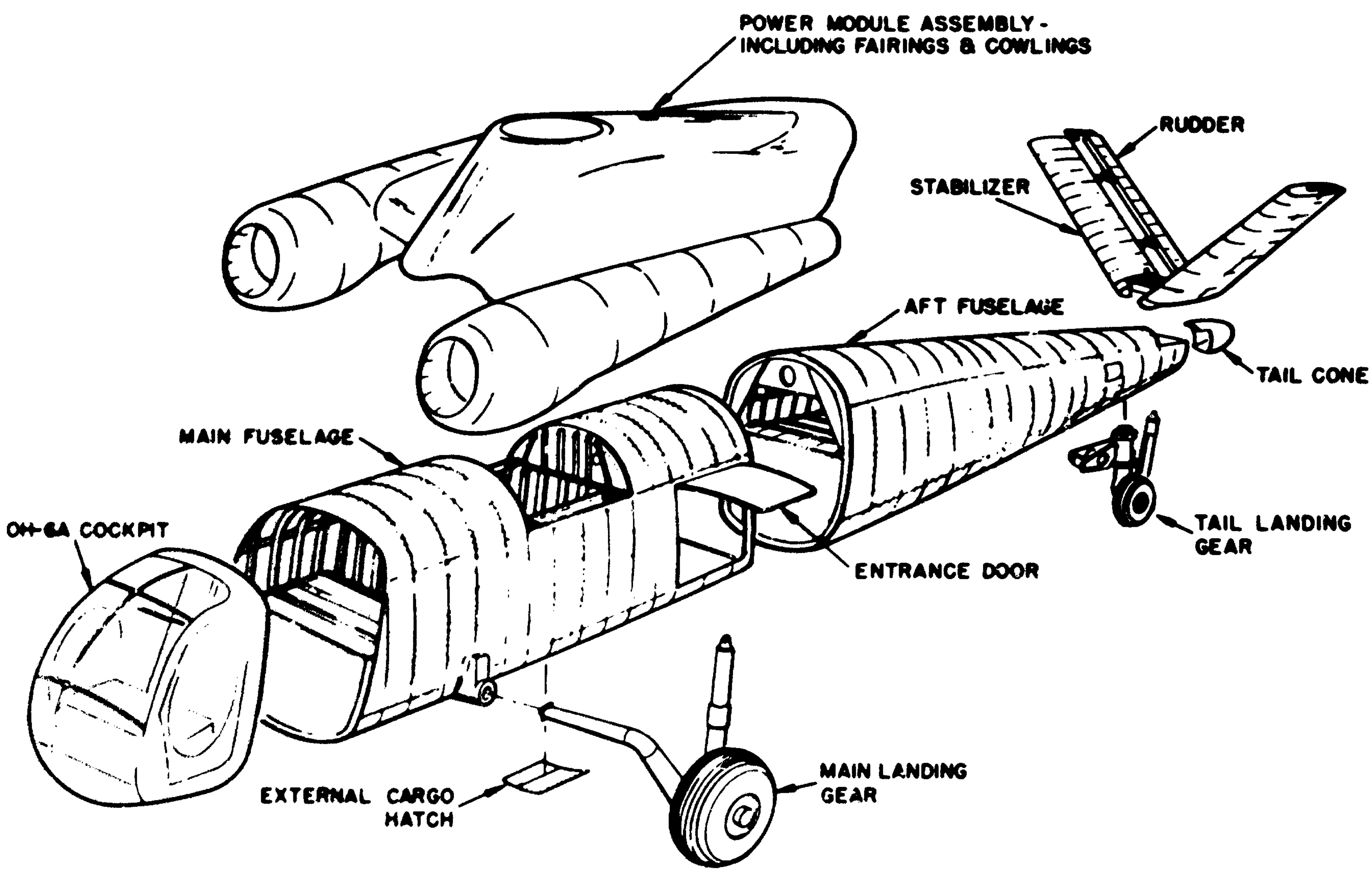


Figure 9. XV-9A Structural Components.

The main fuselage of the XV-9A extends from the cockpit aft approximately 15 feet, and is of conventional semimonocoque construction, employing four longerons, closely spaced frames, and stressed skin, all of aluminum alloy. Special frames are provided at all concentrated load points. Structurally, this main section provides for the attachment of the power module, main landing gear, and forward fuel cell, and contains the instrumentation compartment.

The aft fuselage is a tapered conical section that extends from the side entrance door aft to the tail cone fairing. This section contains the aft fuel cell, yaw-control valve and ducting, tailwheel, and empennage attachments, and is made up of a continuation of the four longerons, frames, and skin. Bulkheads are provided for the attachment of the tailwheel and empennage.

The empennage assembly is a V-tail configuration consisting of stabilizers with aerodynamically and dynamically balanced rudder-control surfaces. A short center section connects the two stabilizers and provides for attachment to the fuselage. The forward attachment is pinned, and the use of adjustable links at the aft attachment provides for varying the incidence 10 degrees.

The main landing gear installation consists of CH-34A components. The length of the main gear oleo strut, attached to the power module front spar, was modified to adapt it for use on the XV-9A aircraft. The CH-34A tail gear assembly is fully castering for ground operation and can be locked for flight.

POWER MODULE STRUCTURE

The power module structural assembly (Figure 10) consists of the nacelles, the horizontal pylon, and the vertical pylon. The complete assembly is bolted to the fuselage at four points (Figure 9). Structurally, this section provides support for the engines, the diverter valves, the hot-gas ducting, the controls, the hydraulic system, and the auxiliary gearbox installation, in addition to providing for the rotor and the main landing gear oleo strut attachments.

The two nacelles, located at the outboard ends of the horizontal pylon, house the engines, the diverter valves, and the tailpipes. The diverter valve is located between the two main nacelle frames, which are integral with the forward and aft spars of the horizontal pylon. Tail cone assemblies cantilever aft from the rear spar frames and provide support and access to the tailpipe.

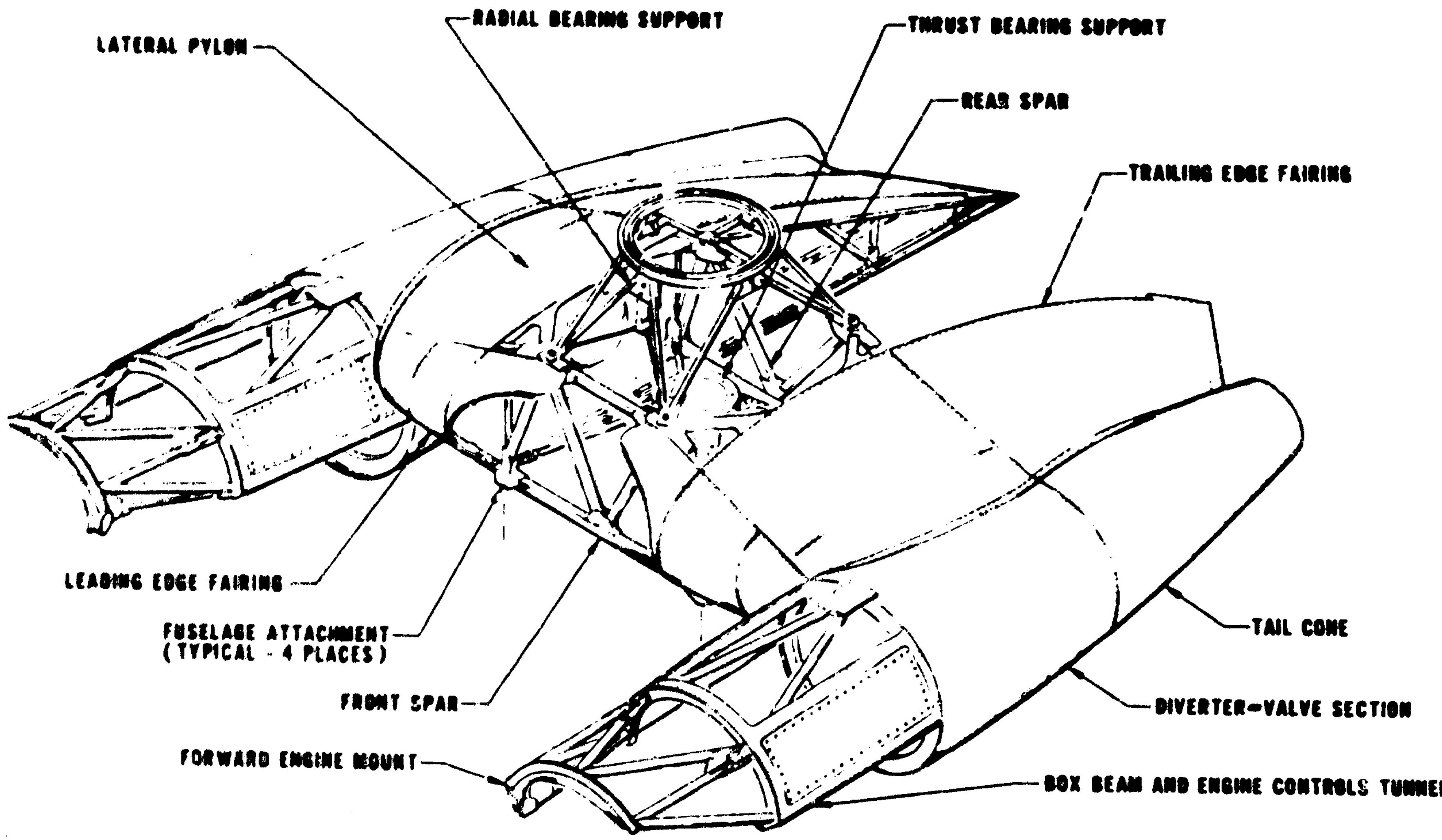


Figure 10. XV-9A Power Module.

The horizontal pylon structure between nacelles provides for the attachment of the rotor-support trusses. The center section also houses the rotor-control cylinders. Spar fittings at the junction of the spars and inboard ribs provide the four attachment points to the fuselage. Non-structural leading and trailing edge fairings complete the horizontal pylon.

The rotor is supported from the horizontal pylon by two welded-steel tube trusses, one supporting the upper radial bearing and the other supporting the thrust bearing at the lower end of the rotor shaft.

FUEL SYSTEM

The airplane fuel system consists of two independent systems, one for each engine, with crossfeed capability between systems. Using the system valving, fuel can be made available to both gas generators from either fuel cell, or to one of the gas generators from both fuel cells. Each system consists of a fuselage-mounted 250-gallon bladder-type fuel cell, boost pump, shutoff valves, strainers, vents, and drains. Shutoff valves are located to give the maximum degree of safety in the event of an emergency. The system is designed so that it can be completely drained.

FLIGHT CONTROL SYSTEM

The flight control system is composed of the rotor control system and the yaw control system.

The rotor control system may be divided into three distinct installations:

1. Pilot linkage (Figure 11)
2. Stationary power linkage (Figure 12)
3. Rotating linkage (Figure 12)

The pilot linkage consists of the cyclic stick and collective pitch lever, their associated mounting structures, bellcranks and push rods, the mixer and its support structure, the artificial "feel" system, and the linkage attached to the power control actuator servo-valve spools.

The stationary power linkage is composed of the hydraulic power control actuators and servo valves; the actuator attaching linkage and support structure, all contained within the central bay of the power module; the stationary swashplate; and the swashplate drag link. The forward (longitudinal) actuator is mounted vertically and attaches to the forward

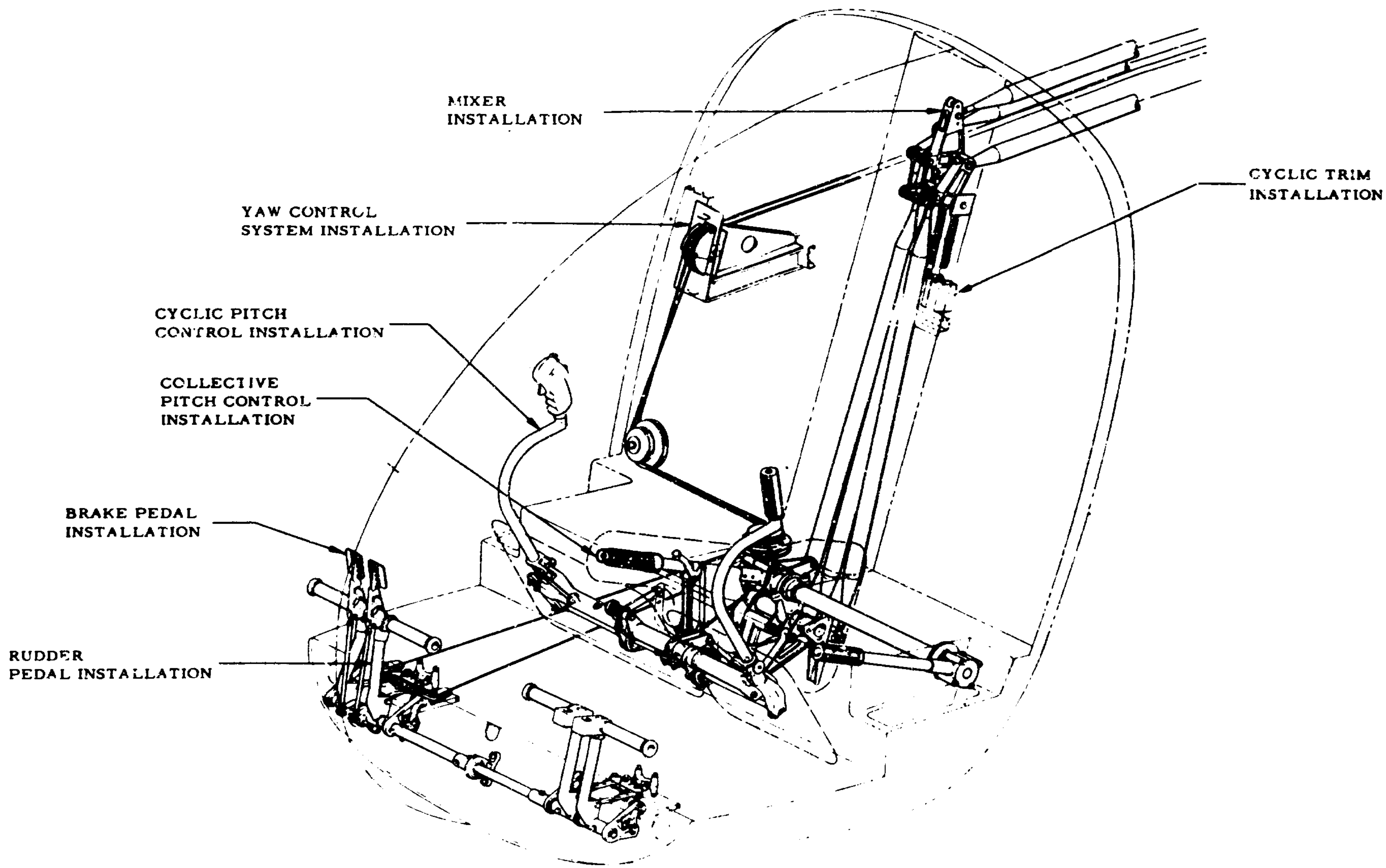


Figure 11. XV-9A Cockpit Flight Control System.

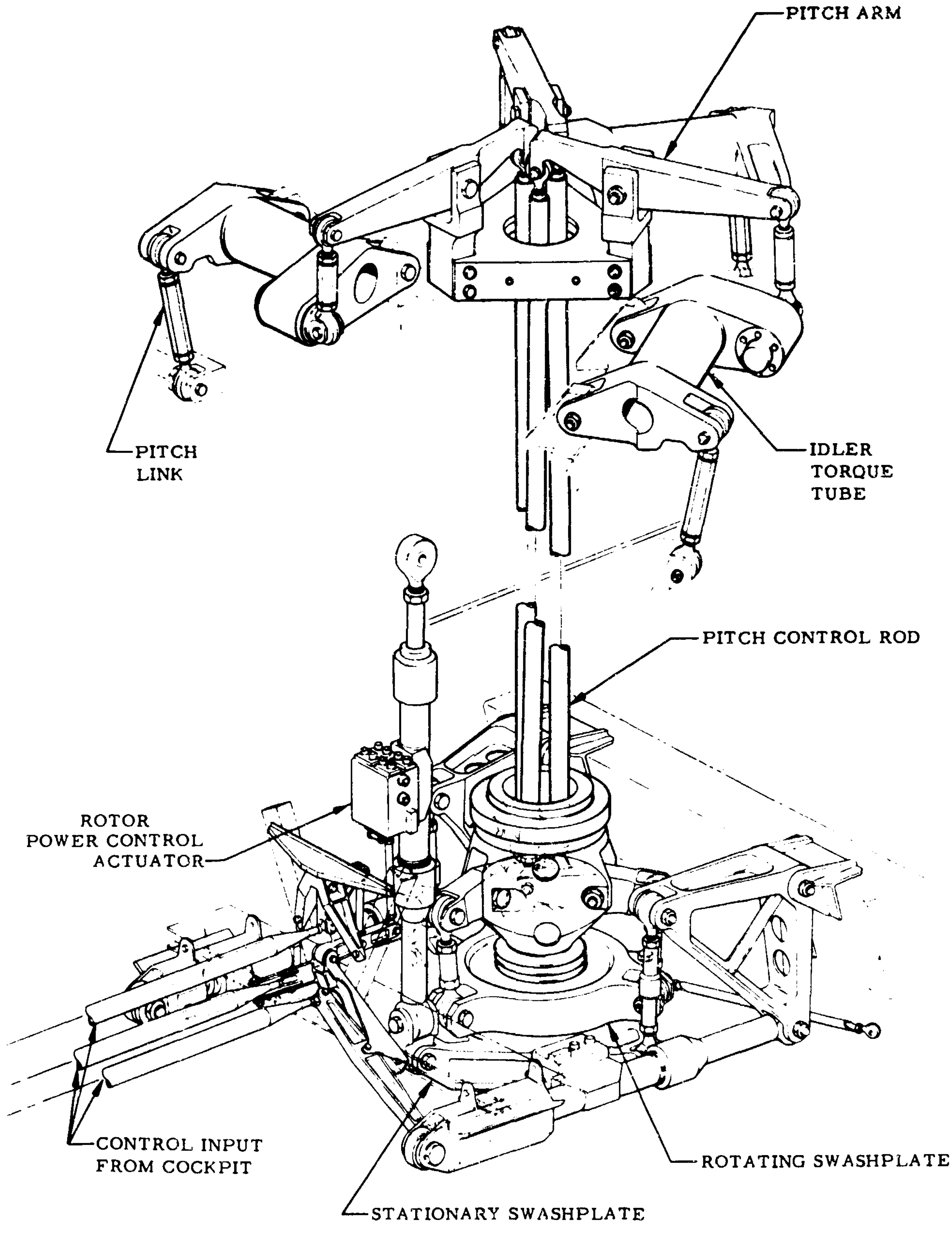


Figure 12. XV-9A Rotor Power Control System.

edge of the swashplate, while the two lateral actuators are mounted horizontally and act on the swashplate through offset bellcranks and push rods. This arrangement allows the use of the standard 90-degree T-arrangement of the stationary swashplate, and at the same time permits all the actuators to be supported entirely within the power module structure.

The rotating linkage consists of the rotating (upper) swashplate, swashplate centering spindle, lower linkage and support housing, the central push rods, upper linkage and support structure, the hub-mounted torque tubes, and the instrumented blade pitch links.

The yaw control system utilizes the rotor-system gas to produce the required yaw force by discharging these gases through variable area nozzles located diametrically opposite each other in the aft fuselage. The system consists of rudder pedals, linkage, and cable system to the yaw valve and rudder. The yaw valve and rudder are linked together and move simultaneously to provide directional control.

HYDRAULIC SYSTEM

Two independent hydraulic systems furnish power for operation of the three tandem-cylinder hydraulic rotor-control actuators and the two diverter-valve actuators. One system furnishes fluid for the turbine speed-governing system. Both systems, primary and utility, are powered by engine-driven variable displacement pumps. The primary system is also powered by the rotor accessory gearbox-driven pump. Both systems operate independently at 3,500 psig during normal operation. In the event of system failure, hydraulic power is automatically switched to the other system. In the event of dual gas-generator failure, the rotor-driven pump will supply the aircraft requirements.

ELECTRICAL SYSTEM

The aircraft electrical system consists of a primary 28-volt d-c system and a secondary 400-cps system.

The 28-volt d-c system is supplied with power by two engine-driven d-c generators and a 24-volt battery operating in parallel. External power is connected to the aircraft through a standard external power receptacle mounted on the left side of the fuselage. Cockpit controls are provided to adjust generator voltage, to open or close generator line and field, and to connect or disconnect battery power or external power.

The secondary electrical system is a 400-cps single-phase system with two subsystems, one at 115 volts and the other at 26 volts. They are used to power synchros, gyros, and portions of the flight test apparatus. Power for the 400-cps system is obtained through a transistorized inverter whose capacity is 250 volt amperes at 115 volts.

COCKPIT ARRANGEMENT

The arrangement of the cockpit instruments and controls is shown in Figure 13. The controls for the most important systems have been grouped along the aircraft centerline for ready access. The fuel-system control panel is located directly aft of the power control quadrants. Alongside the fuel control panel are the hydraulic and gas-system control panels. Adjacent to the power control quadrant are the gas-generator start switches, and forward of these switches are the electrical-system controls and instruments and the aircraft light switches.

A VHF radio is installed and is controlled by a two-position press-to-talk switch incorporated into the cyclic grip.

The flight-instrument installation, using standard aircraft instruments, is not shock mounted. Warning lights are provided on the instrument panel and on the aircraft warning light panel located directly above the instrument panel.

Associated with the warning panel is the emergency panel, located on the center cockpit canopy bow above the pilot. On this panel are the fire extinguishing switches and the crash switch.

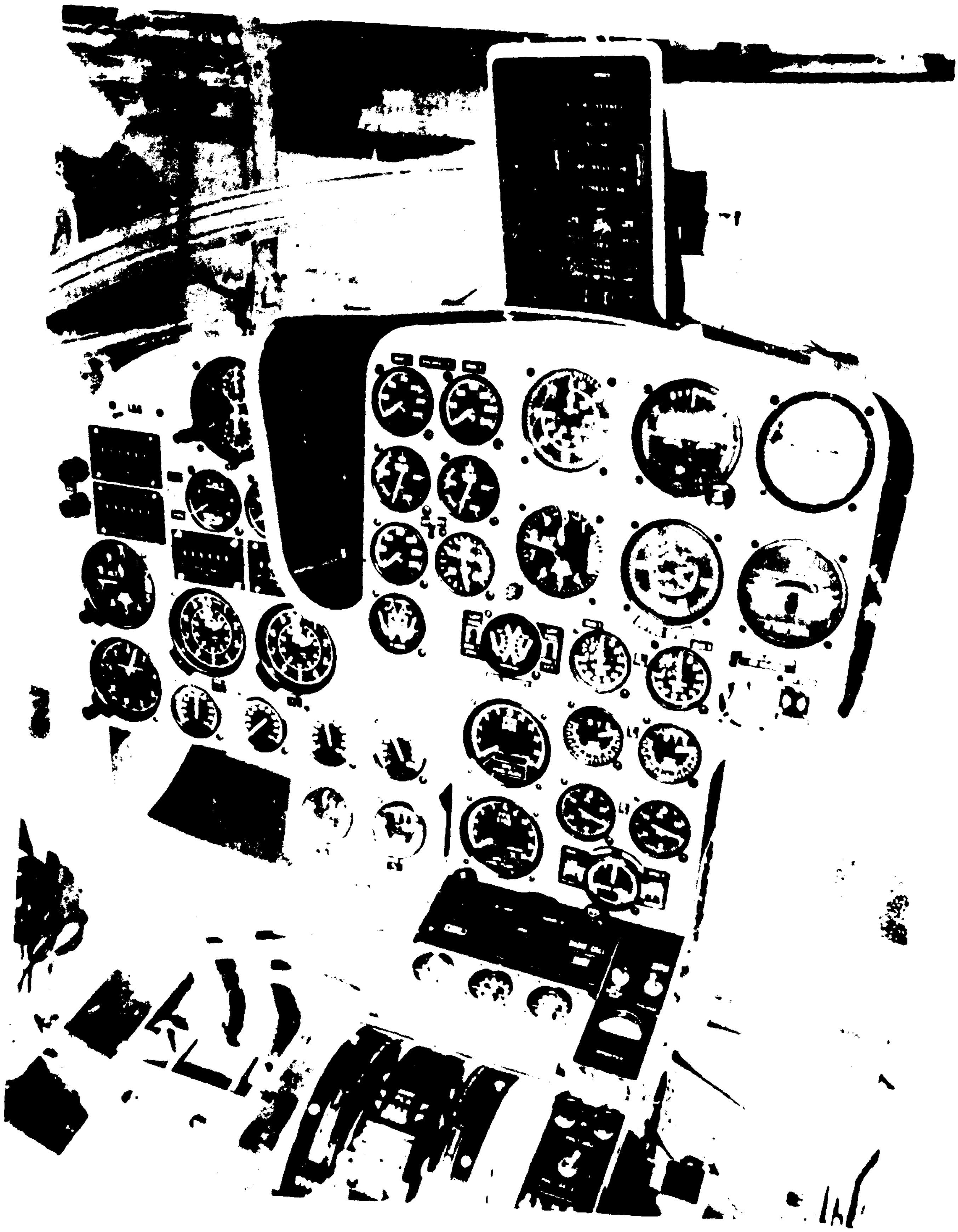


Figure 13. XV-9A Cockpit Instrumentation.

ENGINE AND WHIRL TESTS

DISCUSSION

The first phase of testing consisted of engine tests to establish the operation of two YT-64 gas generators installed in the power module section of the XV-9A Hot Cycle Research Aircraft and to evaluate gas-generator control and performance during twin-engine operation with mixed gas flow through a common exhaust nozzle. These tests are summarized in Table I.

The second phase of testing, consisting of the rotor whirl tests summarized in Table II, accomplished a functional checkout of the complete propulsion system portion of the XV-9A to assure adequacy for flight. These tests also demonstrated compatibility of the rotor and twin YT-64 powerplant operation with associated controls, diverter valves, subsystems, and ducting.

ENGINE TESTS

The basic engine test configuration (Figure 14) consisted of two YT-64 gas generators installed in the XV-9A power module, with the fixed-duct portion of the propulsion system. The power module assembly was installed on top of the whirl tower. Engine controls, instruments, switches, circuit breakers, and test instrumentation were contained in a control van located approximately 75 feet from the base of the tower.

These tests provided a functional checkout and verification of the fixed-duct portion of the XV-9A Hot Cycle Research Aircraft, including J-85 diverter valves, engine diverter-valve seals, transition ducts, and engine tailpipes. Testing consisted of 17 hours 30 minutes of individual engine operation with gas flow through diverter valves in overboard position and 7 hours 48 minutes of twin-engine operation with mixed gas flow through the common exhaust duct and exit nozzle.

The engine and power module tests were conducted prior to the installation of the rotor in the propulsion system. The following test objectives were accomplished:

1. Definition of engine operating line
2. Verification of engine manufacturer's test data
3. Selection of topping procedure and topping conditions for each engine

TABLE I
ENGINE TEST SUMMARY

			Operating Times	
Number		Purpose/Objective	Engine 1	Engine 2
			S/N 250010-4	S/N 250013-5
1	10-28-63	Engine and systems shakedown, Engine 1	00:22	-
2	10-29-63	Engine and systems shakedown, Engine 2	-	00:38
3	10-31-63	Steady-state operating data, Engine 2	-	1:02
4	11-1-63	Steady-state operating data, Engine 1	00:49	-
5	11-4-63	Engine 1 topping check and single-engine transients; twin-engine operation idle; diverter valve functional checkout	1:24	00:21
6	11-5-63	Engine 2 topping check and single-engine transients; twin-engine steady-state operating data; (engine 013-5 failure occurred)	00:4	1:24
				*Total 3:25
7	11-21-63	Airflow calibration, Engine 1; single-engine transients	1:40	
				S/N 250026-1A
8	11-23-63	Engine shakedown, Engine 2		00:39
9	11-26-63	Steady-state operating data, Engine 2	-	00:52
10	11-27-63	Steady-state operating data, Engine 2	-	1:07
11	11-29-63	Engine 1 topping check; variable geometry schedule check, Engine 2; effect of tailpipe tabbing, Engine 2 (5-sq in. tabs added)	00:39	00:58
12	12-2-63	Airflow calibration Engine 1	1:40	-
13	12-3-63	Engine 2 topping check; variable geometry schedule check; airflow calibration data	-	1:49
14	12-5-63	Twin-engine**, steady-state operating data (exhaust nozzle area = 109.6 sq in.)	1:23	1:08
15	12-6-63	Twin-engine**, steady-state operating data; twin-engine transients and mismatch (exit area = 109.6 sq in.)	3:20	3:07
16	12-7-63	Rapid twin-engine** transients and mismatched accelerations (exit area = 109.6 sq in.)	2:11	2:05
17	12-10-63	Twin-engine**, steady-state data with reduced exit area; rapid twin-engine transients with reduced exit area; HTC inlet airflow data for calibration (exit area = 96.6 sq in.)	1:57	1:51
		TOTAL	16:05	13:36

*Engine S/N 250013-5 removed prior to run 7; Engine S/N 250026-1A installed.
**Gas flow through diverter valves and common exhaust nozzle.

TABLE II
WHIRL TEST SUMMARY

Run Number	Date	Purpose/Objective	Operating Times: Engine 1 (S/N 250010-4)	Operating Times: Engine 2 (S/N 250026-1A)	Operating Times: Rotor
18	3-24-64	Engine and systems shakedown	00:32	00:31	-
19	3-27-64	Rotor shakedown (run aborted because of rotor oil leak)	00:38	00:33	00:01
20	4-2-64	Rotor and systems shakedown	1:01	00:52	00:23
21	4-3-64	Rotor-speed buildup and rotor tracking checks; rotor thrust buildup and performance (θ = 4°, 6°, and 8°)	2:04	2:00	1:52
22	4-7-64	Rotor thrust buildup and performance (θ = 10° and 11°), simulated emergency - rapid rotor shutdown	00:55	00:53	00:38
23	4-9-64	Rotor performance (θ = 11° and 12°)	2:19	1:42	1:00
24	4-22-64	Yaw-valve functional checkout and thrust measurement; cyclic control response and blade dynamic characteristics, diverter-valve leakage measurement	1:55	1:51	1:21
25	4-27-64	Yaw-valve thrust and power measurements; engine and rotor sound spectral data	2:32	2:25	1:39
26	5-1-64	Rotor performance (θ =7°); diverter-valve leakage measurements; engine nacelle dynamic characteristics	1:34	1:29	1:29
27	5-7-64	Triduct displacement (run aborted because of ground equipment failure)	00:23	00:18	00:04
28	5-8-64	Blade-tip closure valve functional checkout; blade dynamic characteristics with simulated mass of new tip cascades; diverter-valve leakage measurements; cyclic control reversals	2:00	1:58	1:38
29	5-9-64	Engine topping adjustments, pilot familiarization (run aborted because of loose tape on blade leading edge)	00:55	00:50	00:02
30	5-11-64	Blade-tip closure valve functional checkout; cyclic control reversals; collective control reversals; rotor downwash data; Y-duct crossflow vane data	3:16	3:04	2:30
31	5-14-64	Rotor performance, maximum topping, sealed leading edge segments and cleaned compressors, θ = 3°, 5°, 7°, 9°, 10°, 11°, 12°, collective-control reversals, cyclic-control reversals, rotor acceleration characteristics; Y-duct crossflow vane data; simulated emergencies	4:38	4:16	3:39
32	5-18-64	Structural frequency investigation, d-c generator regulation and adjustment, revised rotor lube system checkout	00:51	00:48	00:32
		TOTAL	41:33	37:06	15:49

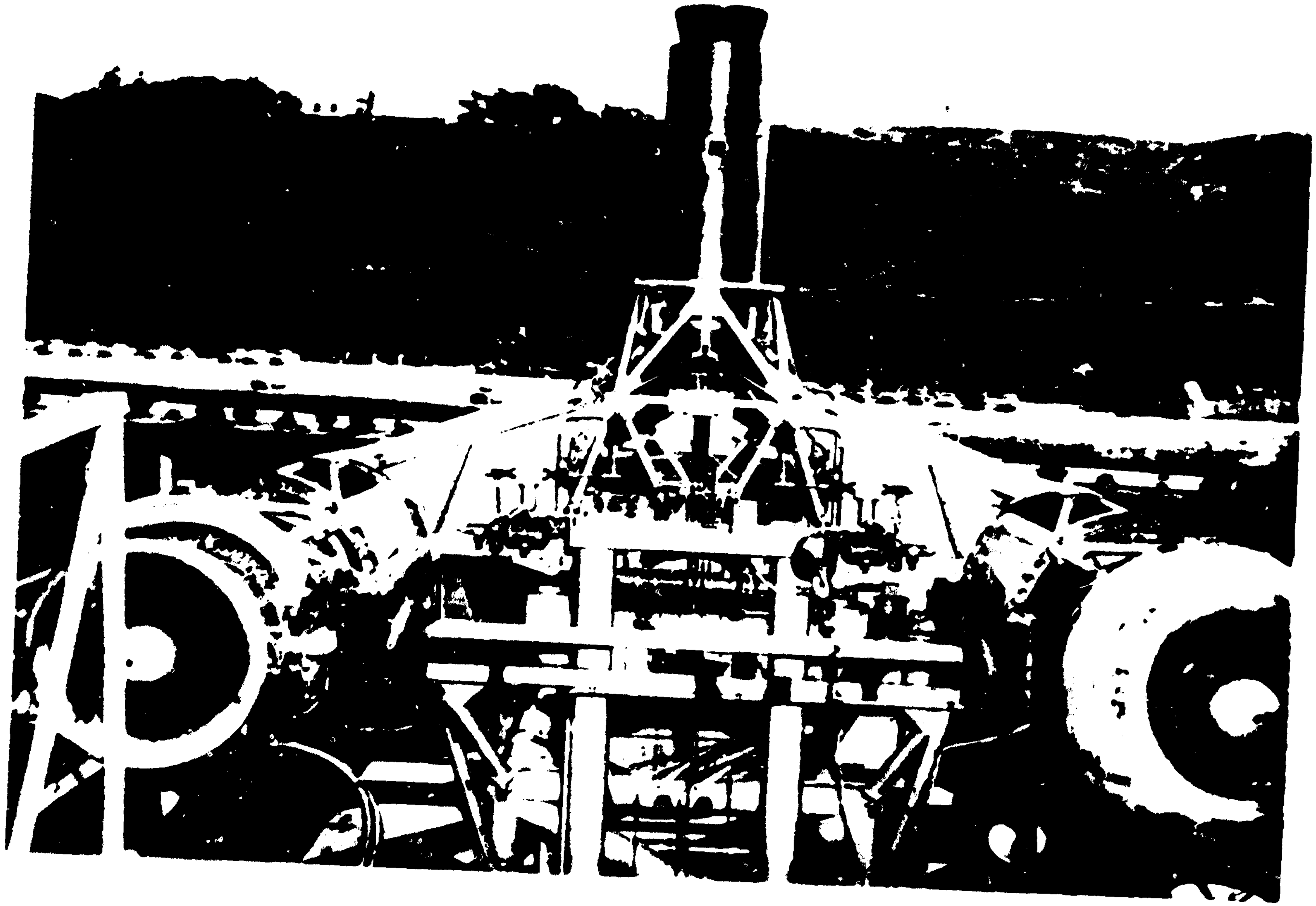


Figure 14. XV-9A Twin-Engine Test.

4. Twin-engine matched steady-state operation
5. Twin-engine matched transient operation
6. Mismatched engines in steady-state and transient operation
7. Simulation of engine or diverter-valve failures
8. Comparison of various engine-flow measuring methods
9. Measurement of pressure loss in the duct system

More detailed information relative to performance operation may be found in Reference 2.

Prior to twin-engine operation of the YT-64 gas generators with mixed exhaust flow, each engine was run individually with its diverter valve in the overboard position to obtain the engine operating line data.

The changeover from individual to twin-engine operation was accomplished by stabilizing both engines at idle with diverter valves in overboard position and then actuating the diverter-valve switches simultaneously to rotor position. The exit area of the common exhaust nozzle was set for twice the nominal single-engine exit area.

Diverter-valve actuating time was approximately 0.5 second from overboard to rotor position. Engine operation was completely stable during the transition, and the only perceptible change in the engine instruments was an equalization of P_{T_5} on both engines.

Engine response during twin-engine accelerations and power lever changes was essentially the same as for single-engine operation, and no difficulty was encountered in controlling both gas generators by manual power lever manipulations.

Twin-engine shutdown was accomplished by reducing both engines to idle power and simultaneously actuating the diverter-valve switches to overboard position. Individual engine shutdown was then made by moving the individual power levers to "off" position.

WHIRL TESTS

The basic whirl-test configuration consisted of the power module and rotor assembly installed on the whirl tower, as shown by Figure 15. The rotor was located approximately 30 feet above ground level. The support structure for the power module incorporated four strain-gage load cells used for rotor-thrust measurement.

Extensive instrumentation was installed on the rotor, controls, gas generators, and supporting structure, and quantitative data were obtained

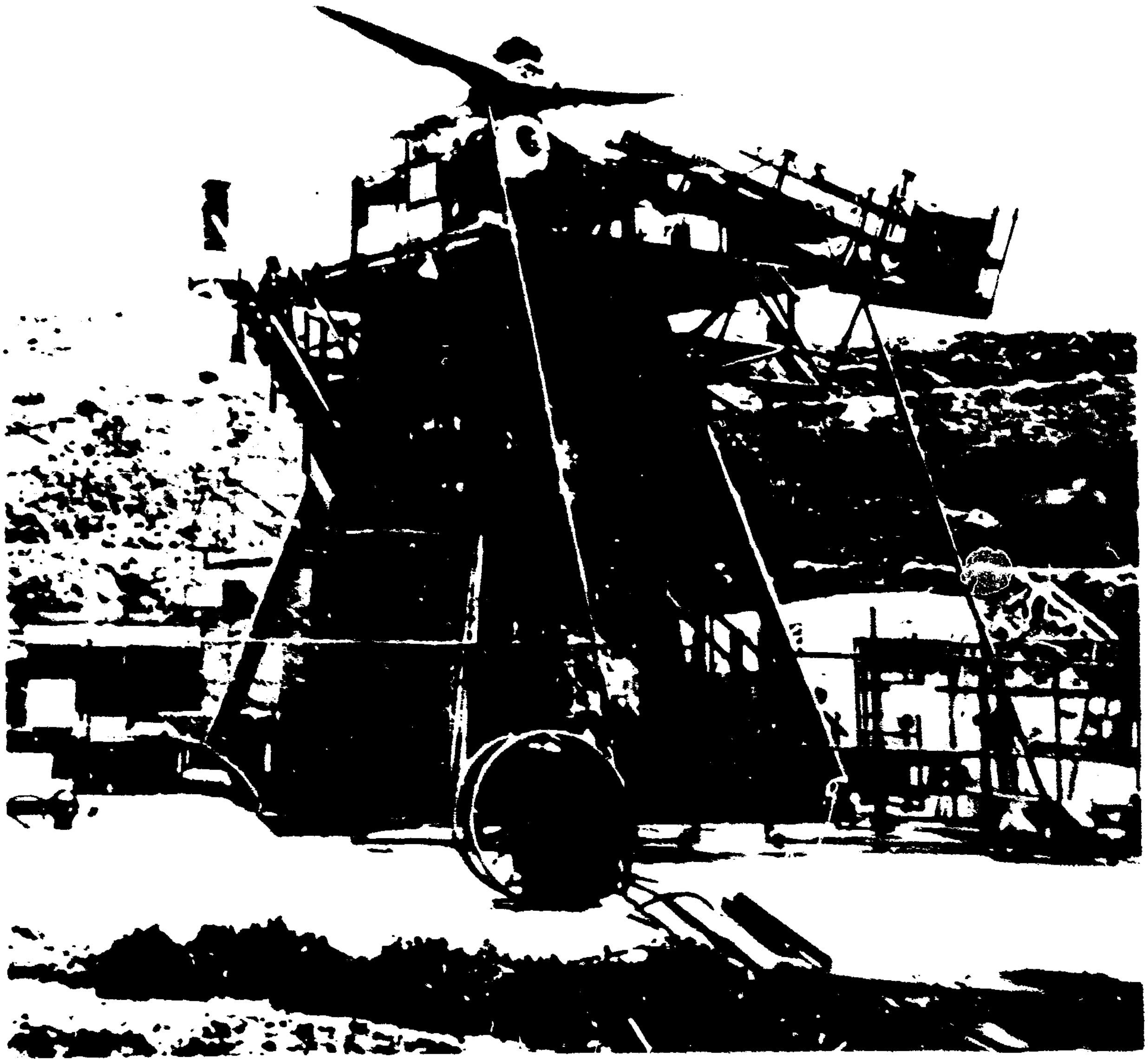


Figure 15. Hot Cycle Rotor During Whirl Test.

from these tests to evaluate engine and rotor performance, rotor-dynamic characteristics, structural loads and temperatures, rotor-control characteristics, yaw-valve operation, engine and rotor sound characteristics, and rotor downwash velocities.

The rotor control system included the flight-type servo actuators. The rotor cyclic and collective pitch controls were located in the control van and were connected to the servo actuators by a cable and pushrod system. Dual engine-driven hydraulic systems provided the power for operation of the rotor controls and diverter valves. Generators were installed on each engine for evaluation of the generator control and regulation system.

Whirl testing was conducted for a total of 15 hours 49 minutes of rotor operation. The yaw-control valve and the two-position blade-tip closure valve were also evaluated during the whirl tests.

The following items of test information were obtained during the whirl-test phase of the program:

1. Rotor performance
2. Rotor system leakage
3. Structural temperatures
4. Structural loads
5. Rotor dynamics
6. Control system characteristics
7. Rotor downwash
8. Rotor-engine sound characteristics
9. Simulated emergencies

A continuous record of temperature distribution in the rotor and propulsion system was kept throughout the test program. No component temperatures over design limits were encountered.

Load measurements were taken during the whirl-test program and were examined for critical loading of the various components.

During the whirl tests, cyclic control pulses of varying severity were applied at different rotor rpm's. In addition, a slow-rpm sweep was made, to locate resonant points at reduced rotor speeds. Results of these tests indicated that the first mode chordwise natural frequency was 1.43 per rev. This frequency was higher than the 1.25 per rev from the original whirl tests, and resulted in lower response to 1 per rev excitation and lower blade stresses than those previously measured on the blade under similar conditions. This higher natural frequency was accomplished by increasing the stiffness of the spars and retention straps.

Evaluation of control response, rotor stresses, and loads in control system components was obtained for transient conditions by applying rapid cyclic control reversals. Cyclic control operation was smooth and stable, with negligible control forces. Evaluation of control response, rotor stresses, and loads in the control system components was also obtained for rapid collective power transients. Again control response was smooth, and all structural loads were within allowable limits.

Rotor downwash velocities were measured using a downwash velocity boom and a movable probe with a standard pitot head approximately 9 feet below the plane of the rotor.

Sound spectral data were taken from the base of the tower at distances of from 100 feet to 400 feet in 100-foot intervals. The highest sound pressure levels were recorded directly in front of the power module and engine inlets. The overall sound level was measured at 112 decibels at a point 100 feet directly in front of the power module. Ambient sound level was 80 decibels. At a point 100 feet from and 90 degrees to the right of the power module, the sound level was recorded at 108 decibels. The rotor thrust was approximately 16,000 pounds for these conditions.

Simulated emergency situations accomplished included:

1. Rapid rotor and engine shutdown
2. Single hydraulic system and control operation
3. Simulated engine failure and engine isolation
4. Engine shutdown by fuel system firewall shutoff valve

The rapid shutdown was initiated with both engines at or near 100-percent compressor speed, with the rotor at 97.5-percent speed, and with a lift of 20,800 pounds. The shutdown was accomplished rapidly and smoothly, with no overloads or detrimental effects on engines, rotor, or structure.

The rotor controls were operated through their normal range and rates with one hydraulic system at zero pressure. The procedure was to bypass each hydraulic system one at a time and to operate cyclic and collective controls, with the remaining system providing hydraulic power at 3,500 psi. Cyclic and collective control operation was smooth and stable while operating on each individual hydraulic system. There were no control transients or noticeable effects when either system was bypassed. Normal operation of both hydraulic systems was restored without any noticeable effect.

The procedure for simulating engine failure was to reduce power on Engine 1 and simultaneously increase power on Engine 2 until the rotor speed dropped to 81 percent. The Engine 1 diverter valve was then actuated to overboard position, which isolated this engine from the system. This operation was accomplished without adverse effect on either engine. The power on Engine 2 was further increased to maximum compressor speed, and rotor operation continued at reduced lift with this engine operating at twice its normal exit area.

Both engines were shut down from idle condition by closing the firewall shutoff valves. Shutdowns were completely normal.

Following completion of whirl testing, the power module and the rotor were removed from the test tower and returned to the factory for tear-down inspection, reassembly, installation of flight-qualified engines, and final mating with the XV-9A fuselage.

TEST RESULTS

The test objectives were completed with very satisfactory results.

The engine and whirl test results can be summarized as follows:

1. The two YT-64 ground-test gas generators with mixed exhaust flow were successfully operated at a wide range of steady-state and transient conditions.

2. The speed-power control characteristics of the Hot Cycle rotor and twin YT-64 gas generators were found to be satisfactory for the complete operating envelope utilizing the standard YT-64 fuel controls.

3. The predicted rotor lift and rotor power characteristics of the Hot Cycle rotor and YT-64 gas generators were substantiated by test measurements (see Figure 16). The maximum measured rotor lift was 23,000 pounds for an engine pressure ratio of 2.62.

4. Operating temperatures of the various rotor components were either as predicted or lower than design limit at the maximum gas temperature attained during whirl test.

5. The leakage of the rotor ducts and seals was less than one-fifth of 1 percent of the total flow as measured before and after whirl test.

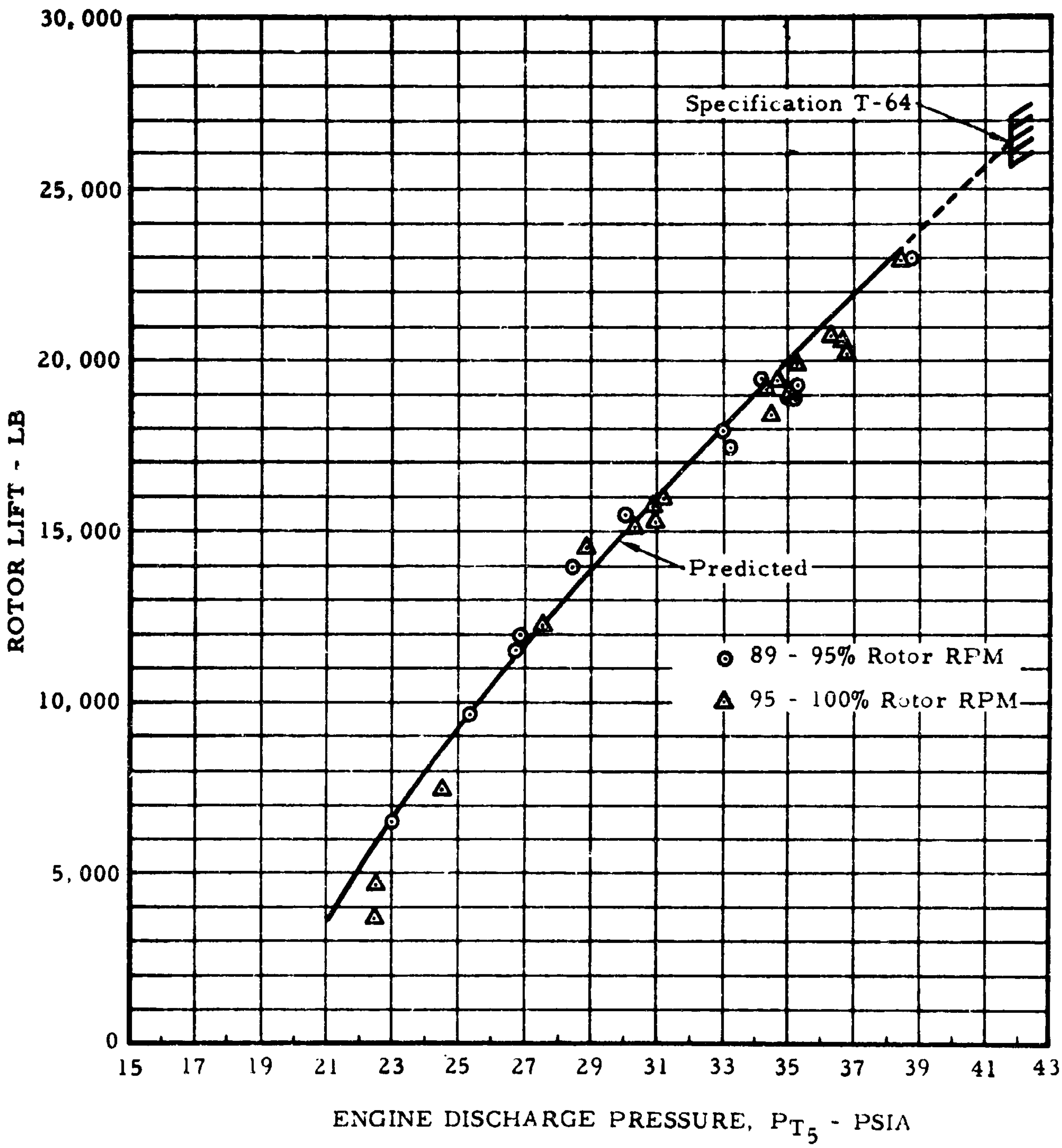


Figure 16. Rotor Lift Versus Engine Discharge Pressure - Whirl Test.

6. Leakage through the J-85 diverter valves amounted to 2-1/2 percent of total engine airflow.

7. Structural load measurements taken during whirl test substantiated the design of the various changes that were incorporated into the Hot Cycle rotor for the XV-9A program. With the exception of blade chordwise moments, all cyclic loads were of moderate magnitude and below endurance limits. Blade chordwise moments were near and slightly above endurance limits during transients at the high rotor lift condition of 23, 000 pounds. There were no blade resonances within the normal rotor operating range.

8. Rotor control response and operating characteristics during all rotor operation, including large cyclic and collective transients and reversals, were determined as satisfactory for flight of the XV-9A.

9. Measured sound levels were approximately the same as had been predicted from previous whirl-test data. The noise level was very comparable to that of a large turboshaft helicopter.

COMPONENT TESTS

DISCUSSION

Tests were conducted on various components of the aircraft to prove the functional and structural adequacy of these components. These tests provided the information necessary for development of the components and furnished data required for rotor system fatigue life calculations. These tests are discussed briefly in the following paragraphs. For additional and more detailed information, refer to Reference 3.

BLADE ROOT-END FATIGUE TEST

Two specimens, each duplicating the XV-9A rotor blade root-end area, were tested under simulated flight loading conditions. The purpose of these tests was to substantiate the structural integrity of the root-end portion of the rotor blade for the planned flight test program.

The test specimens were mounted in the test fixture shown in Figures 17 and 18. This test fixture was capable of applying the static and dynamic loads that the rotor blade is subjected to during flight operation.

A total of 468, 152 cycles, including 32, 100 preliminary cycles at higher load levels, was applied to the first specimen before a crack was detected in the spars. Inspection revealed failure at a bolted connection between the root-end fitting and spar.

After extensive testing on small-scale specimens simulating the spar-to-fitting joint, a fix was made in the spar-to-fitting attachment area, and a second root-end test specimen was fabricated and tested. This specimen was subjected to a total of 413, 000 cycles of loading. At the completion of the 413, 000 cycles, a crack was detected in the front spar at the spar-to-segment attachment bolt at blade station 94. Testing was terminated at this time, although the specimen was still able to support the applied loads. A teardown inspection revealed no distress in the modified spar-to-fitting attachment region.

The calculated rotor-blade service life based on these test results is 107-1/2 hours. A comprehensive discussion of rotor-blade service life is presented in Appendix III of Reference 1. (Further investigations and additional small-scale fatigue tests were conducted during the 20-hour follow-on flight test program. As a result of these investigations, an appreciable increase in service life of the rotor blade is predicted. For a comprehensive discussion of the revised service life calculations, see Reference 8.)

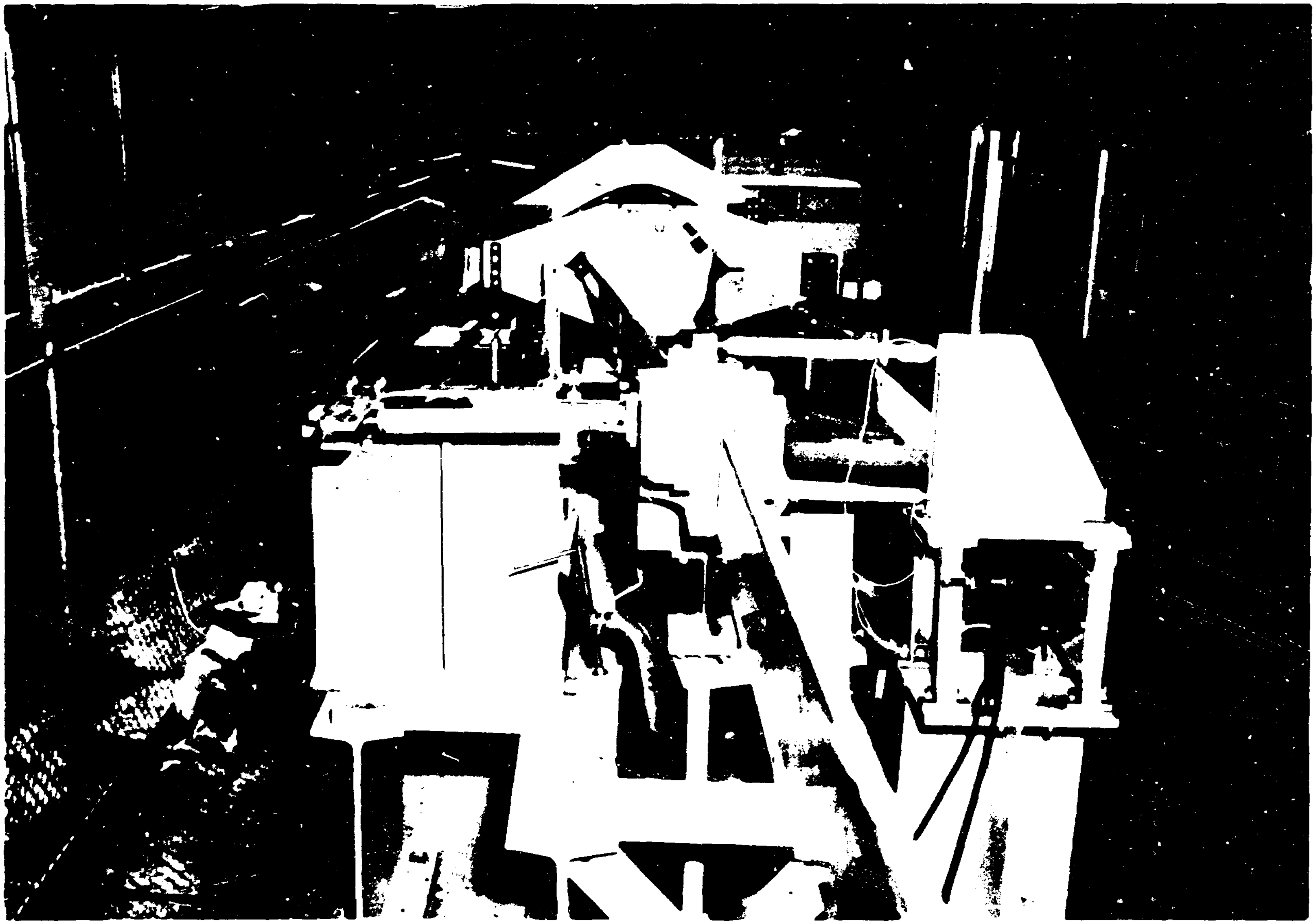


Figure 17. Blade Root-End Fatigue Test Fixture - Chordwise Input Mechanism.

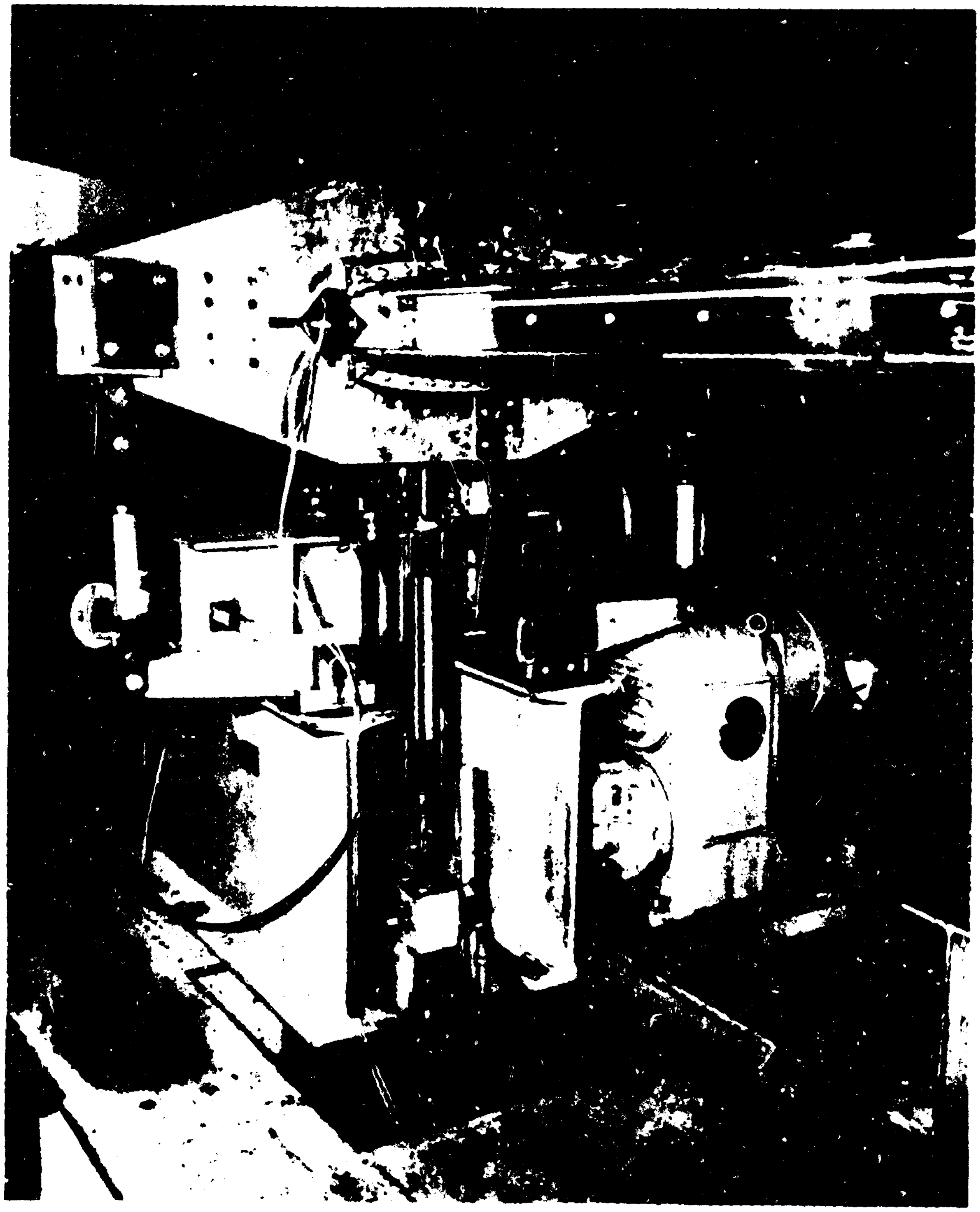


Figure 18. Blade Root-End Fatigue Test Fixture - Drive System.

BLADE CONSTANT-SECTION FATIGUE TEST

The purpose of this test was to substantiate the structural integrity of the constant section of the rotor blade for the planned flight test program. Test conditions simulated the operating conditions of loading, temperature, and pressure.

The test specimen consisted of three segments of a production blade, from station 203.5 to station 241.00. The specimen was installed in the test fixture as shown in Figure 19 and vibrated as a beam under axial tension loading. A spring-loaded exciter induced cyclic flapwise and chordwise bending moments. The axial loading simulated centrifugal force. Tests were conducted with internal duct pressures of 27.5 psig at simulated gas temperatures of 1,200 degrees F.

After completion of the above, the specimen was subjected to additional testing at 125-percent chordwise load, to account for the peak stresses encountered during flight testing.

The specimen initially completed a total of 1,458,000 test cycles. Inspection revealed a 0.15-inch vertical fatigue crack at the spar-to-section attachment at the rear spar station 237.8. This location was in the attachment area of the test fixture and, as such, is not representative of the actual blade. All other areas of the spars were in satisfactory condition, and no evidence of fatigue damage was found in the flexure couplings, the ducts, or the segment areas.

On retest, the specimen completed an additional 25,000 cycles at increased load. Post-test inspection indicated no additional fatigue damage.

REDUCED-SCALE SPAR-TO-SEGMENT AND ROOT-FITTING-TO-SPAR ATTACHMENT FATIGUE TESTS

The purpose of this series of tests was to investigate various reduced-scale configurations simulating improvements to two areas of the blade spar; namely, the spar-to-segment and root-fitting-to-spar attachments. In all, 48 individual specimens were mounted and tested as shown in Figure 20.

As a result of these tests, an improved root-fitting-to-spar attachment configuration was incorporated into the flight rotor-blade spars and the second root-end fatigue test blade.

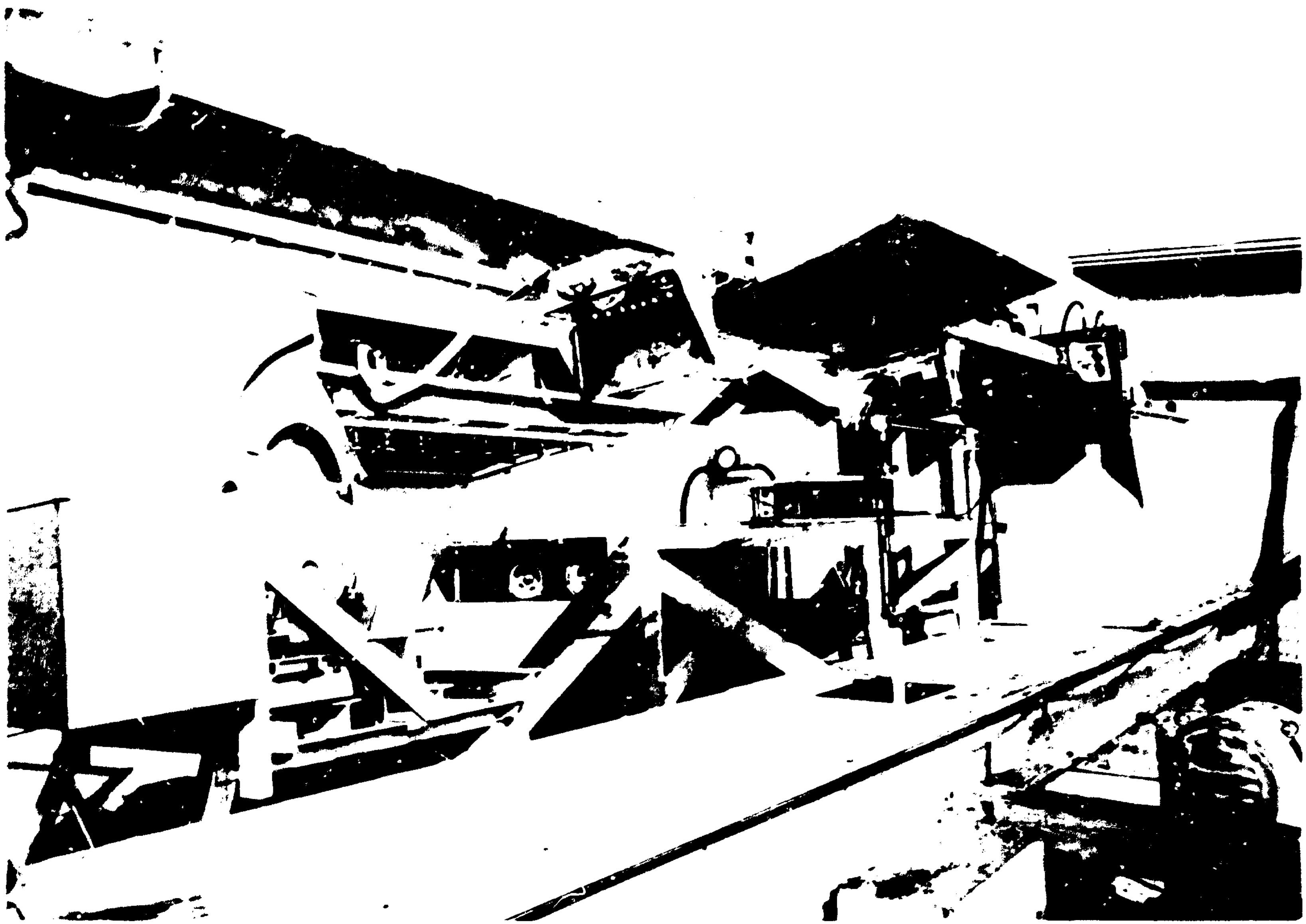


Figure 19. Blade Constant-Section Fatigue Test Specimen and Fixture.

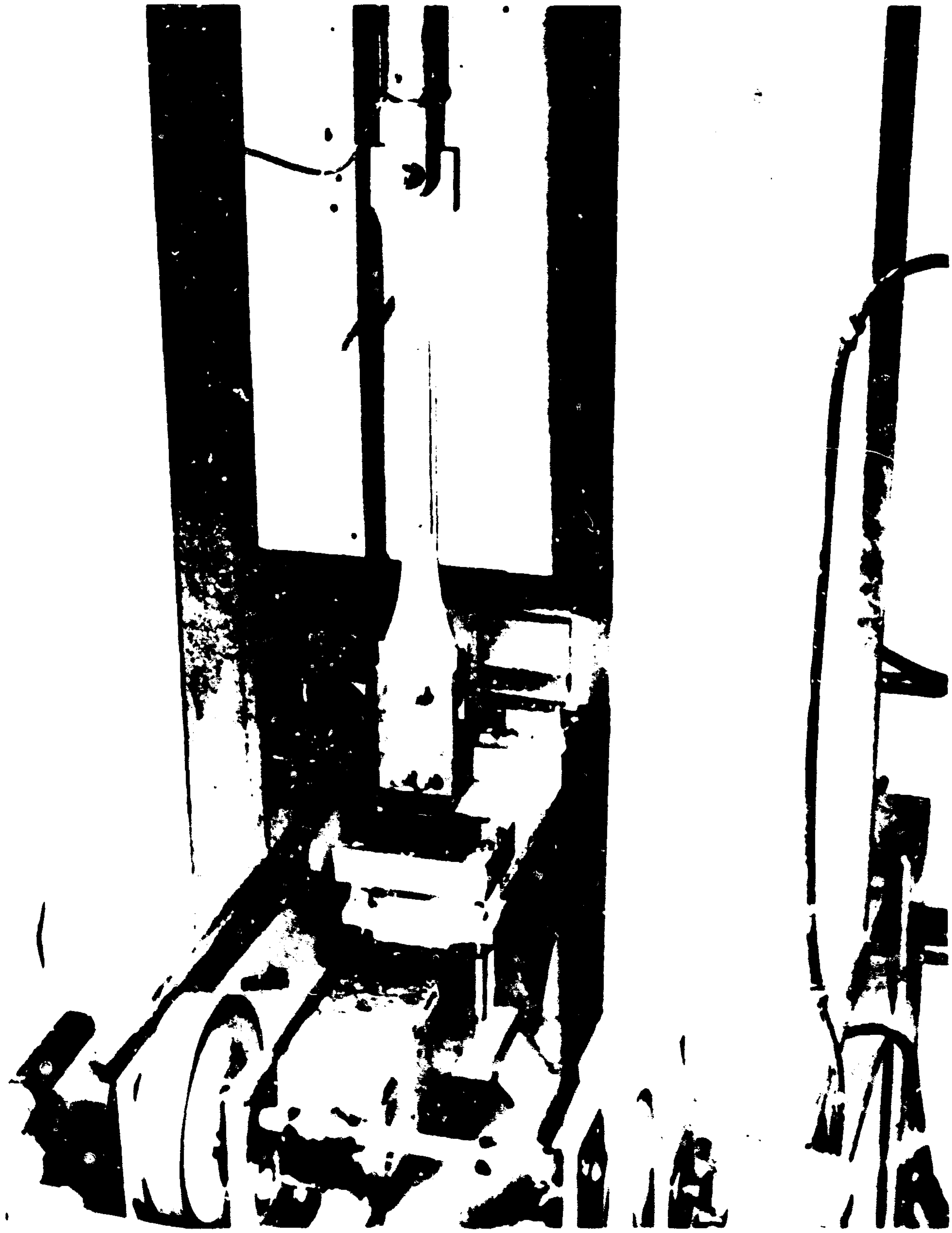


Figure 20. Spar Attachment Fatigue Test.

FATIGUE TEST OF SPAR WITH HOT-GAS IMPINGEMENT

A test specimen simulating a section of the laminated spar was subjected to a flame of 1, 050 degrees F while loaded to represent centrifugal forces and chordwise and flapwise moments. The purpose of the test was to prove the ability of the spar to withstand hot-gas impingement from a ruptured hot-gas duct for a time period sufficient to ensure safe landing of the aircraft.

The test configuration is shown in Figure 21. The specimen successfully completed 33, 130 load cycles — equivalent to a flight time in excess of two hours. Post-test inspection revealed no visible damage to the spar laminations nor to the bonding adhesive.

HUB-GIMBAL SYSTEM FATIGUE TEST

The hub-gimbal bearing and rotor-hub support structure were evaluated under radial and axial thrust loads with oscillating motion.

The bearing and support structure successfully completed 106. 3 hours of test without failure. This test included more than 1, 500, 000 cycles of oscillation and 3, 000, 000 cycles of axial thrust loading with a steady radial load of 11, 470 pounds, an axial thrust of 0 to 4, 170 pounds at 8 cps. and an oscillation of ±10 degrees at 4 cps. Although fretting was evident, the bearing was serviceable at the completion of the tests.

FIXED-DUCT JOINT SEAL TEST

A test configuration duplicating the fixed-duct joints of the root end of the blade was tested under simulated conditions of pressure, temperature, and loading. The purpose of the test was to provide a minimum leakage seal capable of 100 hours of operation.

The configuration using reinforced duct clamps, a metal spacer, and joint sealant compound successfully completed 100 hours of operation under flight-simulated loads of 27. 5-psig duct pressure at 1, 200 degrees F with no measurable leakage.

BLADE-TIP CLOSURE VALVE TEST

A blade-tip cascade assembly was installed and tested on the whirl test stand as shown in Figure 22. The purpose of the test was to verify the functional operation of the blade-tip closure valve under simulated loads and with hot gas supplied by the YT-64 gas generators.

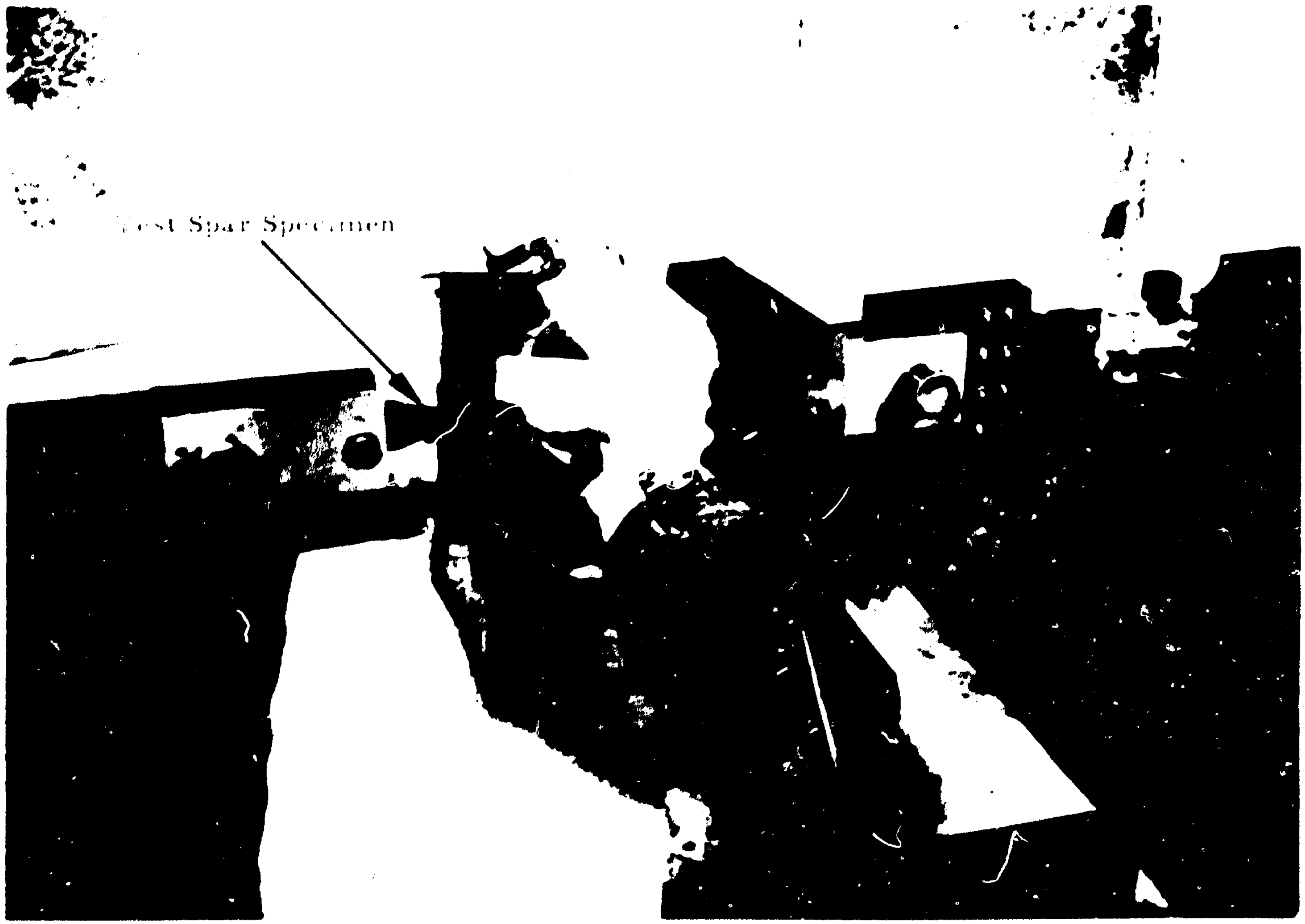


Figure 21. Fatigue Test of Spar Material With Hot-Gas Impingement.

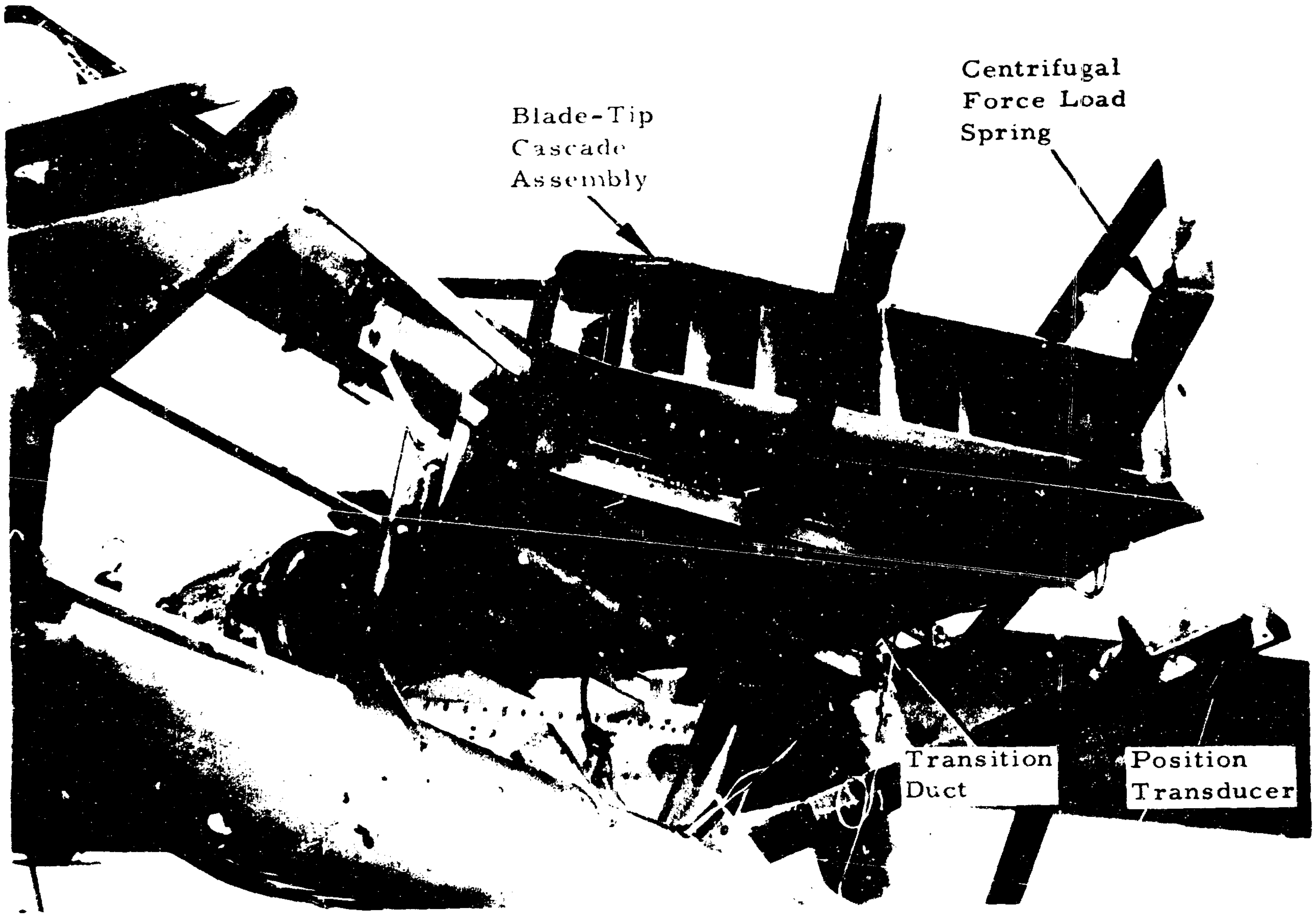


Figure 22. Blade-Tip Closure Valve Functional Test Setup.

A total of 55 cycling functions was performed. The tip closure valve operated satisfactorily with accumulator pressures ranging from 2,800 psig down to 1,000 psig.

ROTOR-SPEED FEEDBACK SYSTEM EVALUATION TESTS

Tests were conducted to evaluate the rotor-speed feedback system (N_f system) designed to drive the YT-64 gas-generator speed-governor shaft (see Figure 23). The system consists of a hydraulic loop between a fixed-displacement pump driven by the rotor and a hydraulic motor at the engine fuel control that drives the governor shaft. The test results showed the N_f system to have a natural frequency approximately three times that required for stable operation, and the system is nearly critically damped. Thus, the system was considered to be satisfactory for use in the XV-9A.

FLIGHT-CONTROL HYDRAULIC-SERVO ASSEMBLY ENDURANCE TEST

An endurance test was conducted on the hydraulic-servo assembly, three of which are used in the flight control system of the XV-9A. The test cylinder was mounted in a harness and was driven and loaded by another hydraulic-servo cylinder (see Figure 24). Approximately 1,000,000 cycles were applied, covering all operating conditions of load and stroke up to relief valve pressures and maximum stroking rates. Test results were very satisfactory and qualified these servo units for flight use.

ADDITIONAL COMPONENT TESTS

Additional tests were conducted on various components of the rotor, aircraft, and systems, as listed below. For details of these tests, see Reference 3.

1. Y-duct and triduct pressure and temperature test
2. Gas-generator diverter-valve seal test
3. Materials evaluation tests
4. Rotor-blade chordwise shear test
5. Rotor-blade spar buckling test
6. Rotor-blade rear-spar proof load test
7. Blade shake test (natural frequencies and mode shapes)
8. Blade static deflection and calibration tests

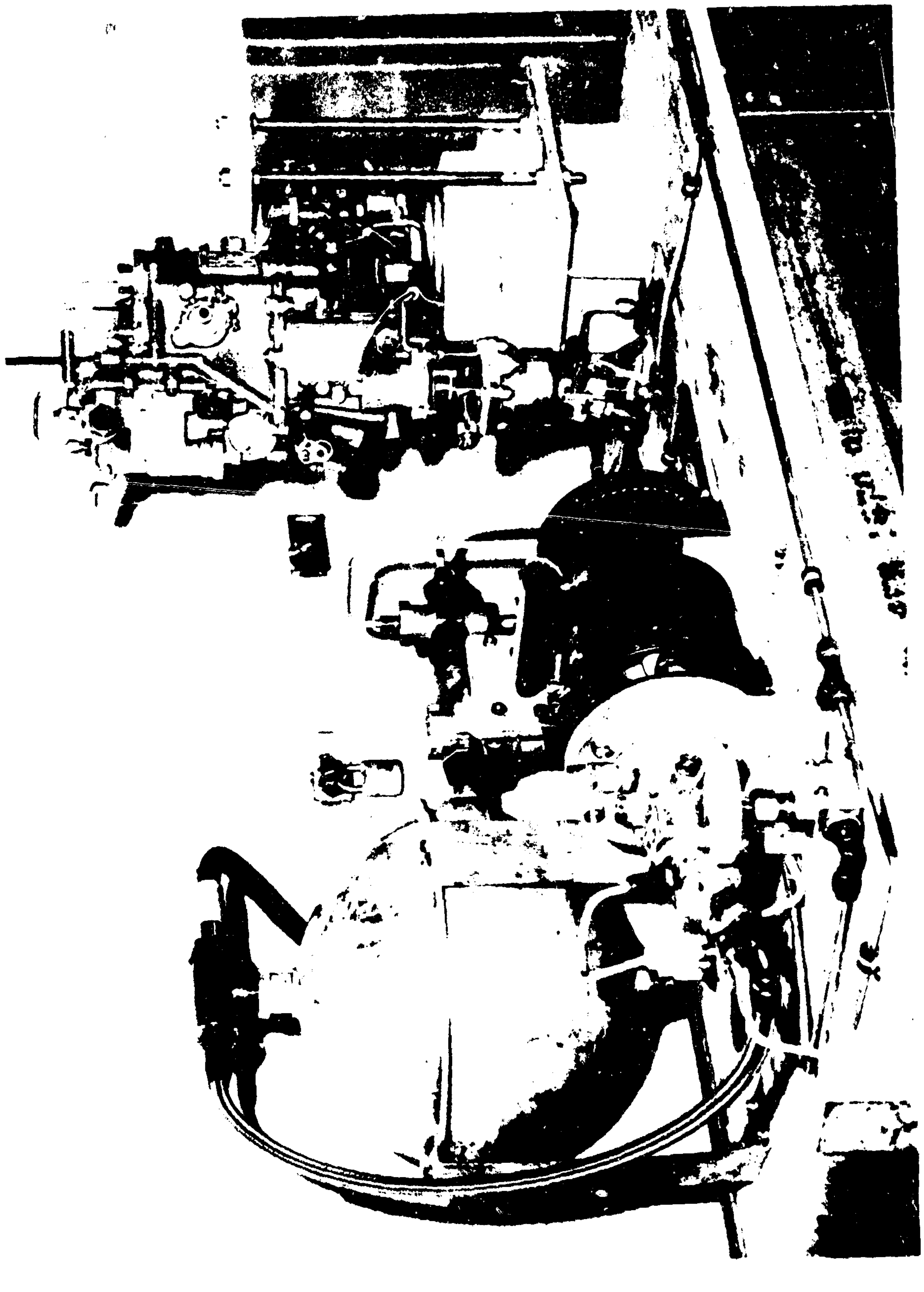


Figure 23. Rotor-Speed Feedback System Evaluation Test.

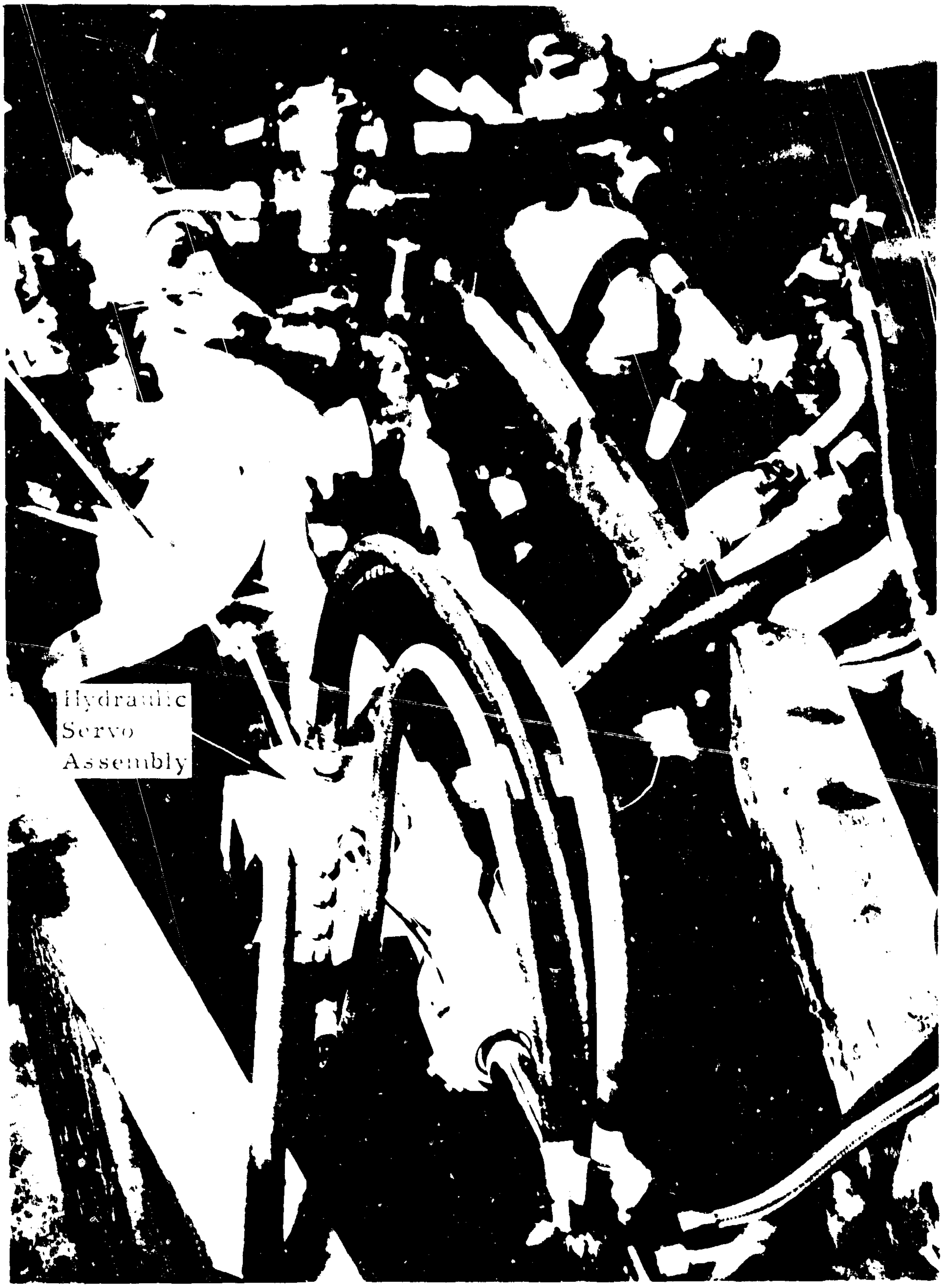


Figure 24. Flight-Control Hydraulic-Servo Assembly Endurance Test.

GROUND TESTS

DISCUSSION

Subsequent to factory completion of the aircraft, a fuselage shake test and preflight and tie-down tests were conducted to provide a functional checkout of the aircraft systems and test instrumentation and a final shakedown of the complete aircraft prior to flight testing.

Prior to preflight tests, a fuselage shake test was conducted with the aircraft supported at the rotor centerline. The purpose of the shake test was to determine the natural frequency of the entire aircraft and its major components. The results and analysis of this test were utilized to determine proper location of fuselage ballast to minimize vibration levels in flight.

The preflight test program consisted of functional checkout of all essential aircraft systems, and included individual system operation through the full operating ranges prior to engine runup. Also, simultaneous operation and evaluation of all systems during engine runup of both YT-64 gas generators was accomplished. The following preflight tests were satisfactorily accomplished:

1. Hydraulic systems – pressure check and functional checkout
2. Flight control system – rigging, functional checkout, and proof loading
3. Engine control system – rigging and functional checkout
4. Rotor lubrication system – functional checkout
5. Electrical system – continuity test and functional checkout
6. Fuel system – pressure test and functional checkout

At the conclusion of satisfactory preflight tests, tie-down testing was initiated. The following tie-down tests were successfully accomplished:

1. Rotor and systems shakedown run
2. Rotor tracking and balance
3. Rotor and systems functional checkout
4. Rotor-airframe vibration investigation
5. Ground instability test
6. Engine nacelle and fuselage cooling checkout
7. Rotor-speed-governing checkout
8. Single-engine rotor operation

The tie-down tests were conducted with the aircraft restrained by cables attached to the upper main landing-gear strut fittings and to the tail gear tie-down fitting. During the latter stages of testing, the cables were slackened to permit limited aircraft lift-off. In all, a total of 13 hours 20 minutes of rotor time was compiled. A summary of tie-down tests is presented in Table III.

TEST RESULTS

All test objectives were completed with very satisfactory results. A brief résumé of the test results is delineated below. For more detailed information relative to tie-down tests, refer to Reference 3.

1. After initial rotor balance and blade tracking adjustments, all rotor operation was relatively free of vibration and dynamic problems.

2. The flight control system, including rotor cyclic and collective pitch controls, yaw control valve, and rudder system, was operated through full operating range during rotor operation at design rotational speed. The only problem encountered involved relatively high pedal forces in the rudder system as a result of aerodynamic spring-back loads developed in the yaw valve. This condition was corrected by a modification of the valve opening, and subsequent operation was satisfactory.

3. Single-engine capability of the Hot Cycle system was satisfactorily demonstrated by actuating one diverter valve to overboard position and closing the blade-tip cascade nozzles to single-engine position with the rotor operating at normal speed. Single-engine power control and governing characteristics were evaluated, and single-engine power available was determined.

4. The rotor governing system, using the N_f power turbine governing function of the YT-64 fuel controls and the hydraulic rotor-speed feedback system, was tested and adjusted for flight. Rotor governing during collective pitch transients equivalent to those required for takeoff to hover and hover to landing was successfully demonstrated.

TABLE III
PREFLIGHT AND TIE-DOWN TEST OPERATIONS SUMMARY

Run Number	Date	Purpose/Objective	Operating Times: Engine 1 (S/N 250027-2A)	Operating Times: Engine 2 (S/N 250101-3A)	Operating Times: Rotor
Engine Run					
1	9-1-64	Engine shakedown and operating data systems checkout	00:39	00:53	-
2	9-11-64	Engine and nacelle cooling checks; engine operational data; fuel system functional check	1:30	00:36	-
Tie-Down Test					
1	9-17-64	Rotor shakedown and tracking	00:44	00:46	00:04
2	9-21-64	Rotor tracking and structural loads	1:01	1:00	00:49
3	9-22-64	Rotor tracking and structural loads	00:43	00:44	00:28
4	9-23-64	Rotor balance investigation	00:32	00:40	00:25
5, 5A	9-23-64	Rotor balance investigation; yaw control valve functional check	1:04	1:02	00:33
6	9-24-64	Rotor balance; engine acceleration characteristics	00:21	00:24	00:14
7	9-25-64	Rotor vibration investigation	00:43	00:45	00:34
8	9-28-64	Rotor vibration investigation; rotor governing checkout	1:34	1:35	1:25
9, 9A	9-29-64	Rotor vibration investigation; rotor governing checkout	01:08	01:06	00:51
10	9-30-64	Rotor-airframe vibration investigation (tie-down cables slack)	00:37	00:39	00:27
11	10-1-64	Tip-cascade system functional checkout; single-engine operating	00:49	00:27	00:36
12	10-9-64	Airframe vibration; engine temperature data; cyclic deflections; generator checkout	00:48	00:48	00:37
13	10-12-64	Airframe vibration; cyclic deflections; diverter-valve hydraulic operation; collective pitch transients	1:15	1:15	1:08
14	10-13-64	Rotor-engine performance; airframe vibration; pitch transients; rotor governing evaluation	1:08	1:10	00:58
15	10-14-64	Ground instability test, aircraft restrained	00:29	00:29	00:20
16	10-19-64	Engine and nacelle cooling; fuselage cooling	1:45	1:46	00:39
17	11-2-64	Functional checkout of reworked diverter valves, engine and nacelle cooling	00:51	00:48	00:16
18	11-3-64	Rotor governing checkout; rotor overspeed system checkout; emergency rotor tachometer checkout	1:41	1:41	1:28
19, 19A	11-4-64	Yaw control valve checkout; rudder-pedal force measurement	2:0[illegible]	1:58	1:28
		TOTAL	21:24	20:32	13:20

FLIGHT TESTS

DISCUSSION

After completion of tie-down tests, the flight test program was initiated. The program was conducted during the period 5 November 1964 through 5 February 1965 and consisted of 21 flights, for a total flight time of 15 hours 42 minutes. All flight test objectives outlined in the flight test program plan (Reference 9) were achieved. The flight test program was completed with no major incidents or delays. The flight and operational characteristics of the XV-9A aircraft and Hot Cycle propulsion system were found to be satisfactory in all flight regimes explored. A summary of flight test operations is shown in Table IV. A comprehensive description of flight testing may be found in Reference 4.

Maintenance of the Hot Cycle rotor system during the flight test program was minimal, and the basic components (including blades, hub, rotating and stationary ducting, seals, bearings, rotor shaft, tip cascades, and rotating controls) were unchanged and required only routine inspection during the program.

Flight plans were prepared in accordance with the flight test program plan and/or as determined by analysis of the data from previous flights and qualitative pilot evaluation.

All flight test operations were preceded by a standard preflight inspection of the airframe, rotor, gas generator, and systems, to ensure safety of flight and proper operation of the aircraft. A preflight inspection and functional checkout of the test instrumentation system was conducted prior to each flight to ensure proper data acquisition.

Two-way radio communication between the XV-9A and ground personnel was maintained during all flight operations for monitoring and coordinating test operations. A flight test log was kept for each flight test operation to document operating time and observations of pilot and test engineer and to facilitate data reduction.

Test data were recorded for evaluation of rotor performance, structural loads and temperature, stability and control characteristics, vibration levels, and aircraft performance for the following conditions:

Engine and rotor start
Rotor acceleration
Taxi and ground handling
Steady hover

TABLE IV
FLIGHT TEST OPERATIONS SUMMARY

Flight Number	Date	Purpose/Objective	Operating Times Engine 1	Engine 2	Rotor	Flight
			S/N 250027-2A	S/N 250101-3A		
1	11-5-64	Taxi and ground handling evaluation; hovering evaluation	1:02	1:03	00:58	00:13
2	11-6-64	Rotor-speed-governing ground checkout; hovering evaluation; transition to forward flight and flare	2:02	2:03	1:42	00:30
3	11-12-64	Hovering evaluation; forward flight to 20 and 30 knots; sideward flight	1:25	1:25	1:13	00:59
4	11-16-64	Hovering evaluation; forward flight to 30 and 40 knots; sideward flight	00:58	1:02	00:39	00:28
5	11-19-64	Hovering evaluation; forward flight to 40 and 50 knots; rudder system force measurement	1:43	1:43	1:26	00:37
[illegible]	11-20-64	Hovering; normal and single hydraulic systems; landing evaluation; manual rotor-speed-governing evaluation	1:32	1:32	1:04	00:33
				*Total 31:38		
				S/N 250026-1B		
7	12-8-64	Hovering evaluation; structural loads and control; forward flight to 70-knot CAS; pacer airspeed calibration (ground run included)	1:56	1:59	1:23	00:23
8	12-9-64	Hovering evaluation; structural loads and control; normal flight pattern; ground run rotor governing check	1:41	1:45	1:31	00:31
9	12-11-64	Hovering evaluation; performance; level flight to 85-knot CAS; pacer airspeed calibration	1:03	1:04	00:57	00:43
10	12-18-64	Hovering evaluation with servo dither; level flight to 95 knots; 200-lb weights on engines; airspeed calibration; ground speed course	1:43	1:43	1:36	1:17
11	12-22-64	Hovering evaluation; 90° turns left and right; forward flight to 100-knot CAS; sideward flight; descents at 50- and 60-knot IAS	1:14	1:14	1:03	00:49
12	12-29-64	Hovering evaluation; sound measurement; level flight data at 50-, 60-, 70-, 80-, and 90-knot IAS; climbs at 102 and 103% N_G; descents at 50-knot IAS	1:31	1:44	1:17	1:10
13	1-6-65	Hovering evaluation; dither on and off; climbs at 40- and 50-knot IAS; level flight turns, $\phi = 20°$, 80- and 90-knot IAS	1:28	1:32	1:13	00:57

TABLE IV (Continued)

Flight Number	Date	Purpose/Objective	Operating Times			
			Engine 1	Engine 2	Rotor	Flight
14	1-11-65	Airspeed calibration; ground course, 80- and 90-knot IAS; climbs at max. coll; level flight turns, $\phi = 30°$; descents $\theta = -2°$	1:26	1:28	1:10	00:56
15	1-14-65	Reset engine idle to 75% N_G; reset governing; hovering performance N_R = 100% and 95%; hover turns, 90° left and right; sideward flight, 20 mph right and left	2:12	2:31	00:42	00:28
16	1-20-65	Engine operating data; cleaned compressors; engine-rotor power transients; climb, 50-knot IAS; level flight turns, $\phi = 30°$; 60-, 70-, and 80-knot IAS	1:28	1:24	1:02	00:29
17	1-21-65	Hovering performance, N_R = 93.5%, 100%, 103%; level flight; speed-power 58-, 71-, 80-, 87-, and 95-knot IAS; rotor downwash; ground run; reset overspeed switch	2:01	2:08	1:55	1:13
18	1-26-65	Hovering performance at 95% and 100% N_R, rotor downwash and sound data; climb, descents, rearward flight; run-on landings	1:48	1:50	1:41	1:10
19	1-26-65	Hovering turns, 180° and 360°; descents at 90% N_R; level flight, 70-knot IAS; tuft photo.	1:51	1:57	1:41	1:04
20	1-28-65	Hover performance point, 101% N_R; hover turns, 180° and 360°; sideward and backward flight; level flight, speed-power, 90-knot IAS	00:53	00:59	00:47	00:33
Ground Run	2-3-65	Engine operating and acceleration data; engine-rotor acceleration data with N_G mismatch; rotor speed vs PLA and rotor speed vs yaw valve characteristics	1:24	1:31	00:52	
Ground Run	2-4-65	Engine rotor acceleration data with mismatch; rotor-engine sound data at 50 ft; single-engine rotor operation and topping; rotor governing with N_f bypass closed	1:58	2:02	1:52	
21	2-5-65	Electrostatic voltage measurement	1:35	1:45	1:28	00:39
		TOTAL	62:06	31:26	45:22	15:42

*Engine S/N 250101-3A removed prior to flight 7; engine S/N 250026-1B installed for flight 7 and subsequent.

Hover turns: 90 degrees, 180 degrees, 360 degrees
Transition to forward flight
Approach to hover
Climb
Level flight
Level flight turns at 20-degree and 30-degree bank angles
Sideward flight to right and left
Rearward flight
Progressive 360-degree turns
Descents, normal and minimum power
Flares to a hover
Landings from a hover and roll-on landings

HOVER PERFORMANCE

On the basis of the relationship of rotor lift to engine pressure ratio, the overall hover performance of the XV-9A agrees very well with whirl test data (Figure 25). The test data also show good agreement with previously predicted performance of the rotor. The capability of the rotor to convert engine discharge pressure into rotor lift is directly dependent upon rotor parameters such as blade duct pressure loss, tip-cascade area and performance, and external blade aerodynamics, but is virtually independent of engine parameters such as compressor bleed and exhaust gas temperature. The data of Figure 25 are plotted without any corrections other than instrument calibrations. The excellent agreement between the test data and the predicted line confirms the overall validity of the performance predictions made for the Hot Cycle rotor system, including agreement of the predicted maximum lift with extrapolation of the test results to maximum T-64 engine discharge pressure.

FORWARD FLIGHT PERFORMANCE

Figure 26 presents a study of forward flight performance of the XV-9A. Data indicated that the parasite drag area, A_{π}, of the helicopter for early flights (up to flight 13) was approximately 62 square feet. During these flights, nacelle fairings were not yet installed, and the yaw valve was apparently open approximately 25 percent. For flights after flight 13, the rudder surfaces were rerigged relative to the yaw valve. Yaw-valve position was then measured and found to be closed in cruising flight, ensuring that a minimum amount of flow bled out the yaw valve. Also, nacelle fairings were installed, producing a drag reduction. Data from flights 13 through 20, corrected for density and weight variations, indicate the parasite drag area to be 45 square feet.

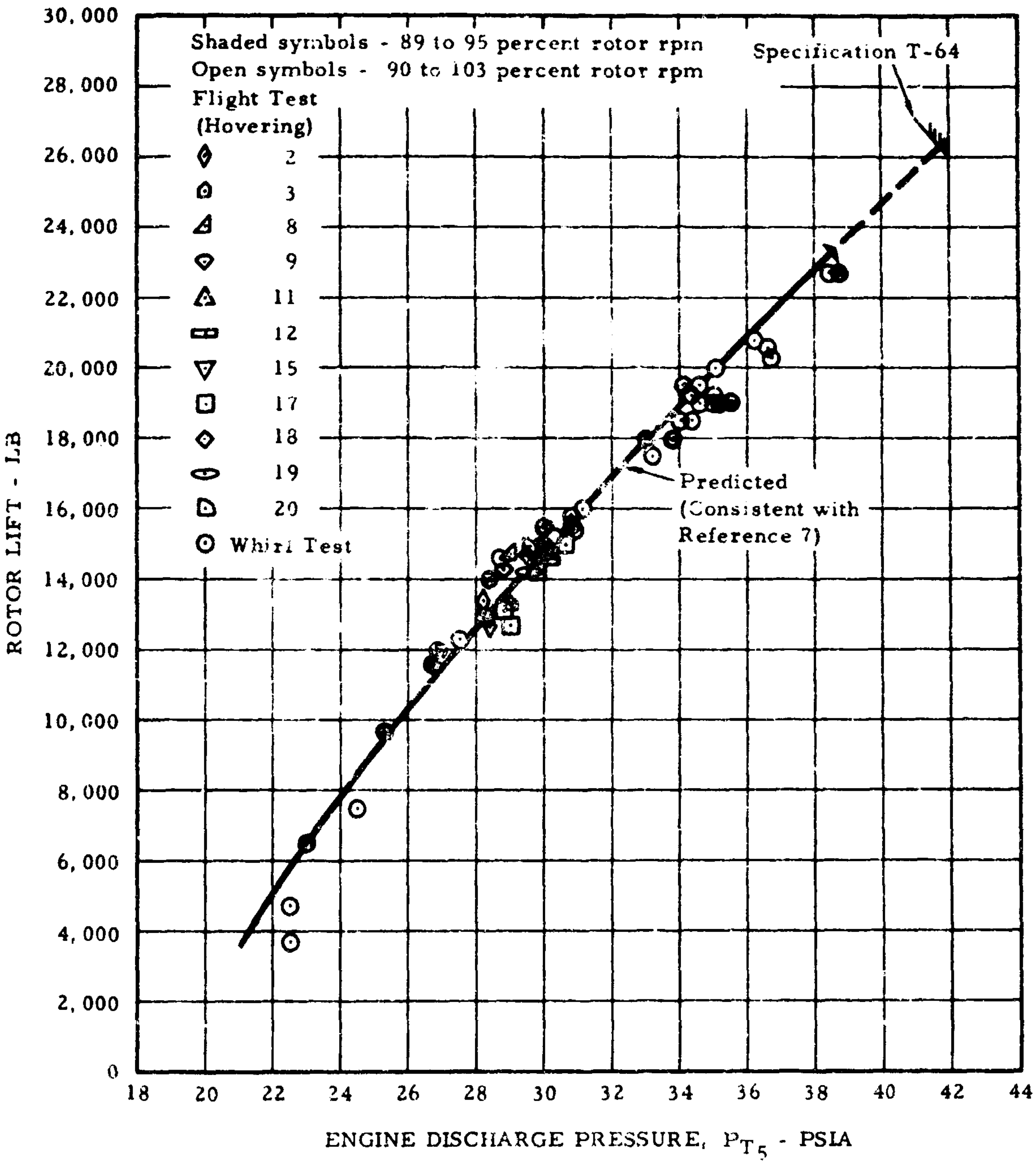


Figure 25. Rotor Lift Versus Engine Discharge Pressure.

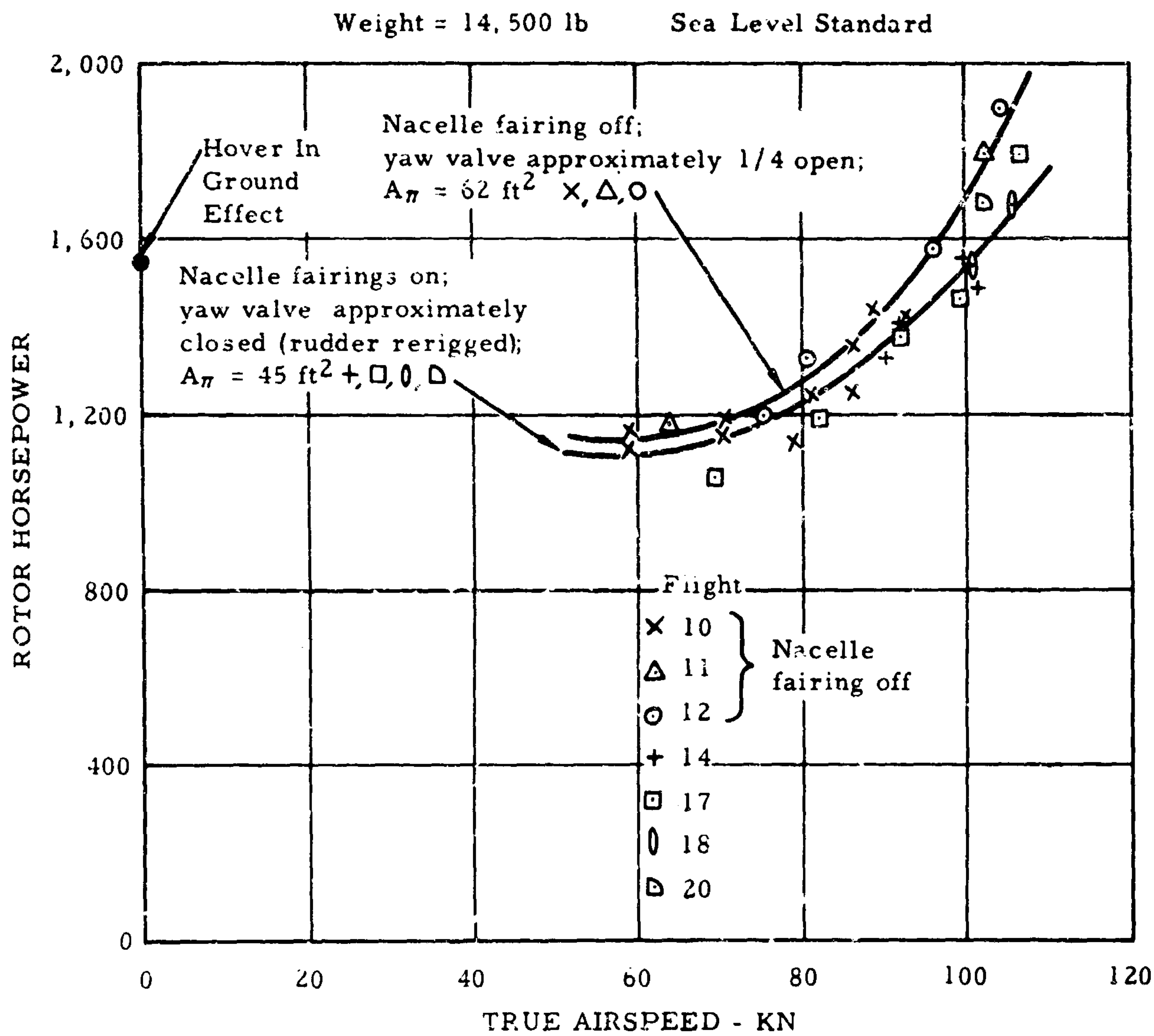


Figure 26. Power Required Versus Airspeed.

The helicopter was then tufted on the upper side of the left horizontal pylon, the left side of the vertical pylon, and the upper left portion of the fuselage back to the tail. In-flight motion pictures indicated that a substantial amount of separation existed, causing the high parasite drag area deduced from flight test. Analysis of the configuration, with information gathered from individual pictures of the tufted helicopter, indicated that the separation was caused by flow interference between the rotor hub and the fixed airframe surface. A component drag analysis was made, and it was concluded that with improved fairings the helicopter parasite drag could be reduced to 22 square feet.

Theoretical calculation of power required, using the reduced parasite drag area of 22 square feet, was made, and the result was plotted, in Figure 27, showing the predicted effect of drag cleanup.

FLYING QUALITIES

The flying qualities of the current XV-9A, in general, were found to be adequate for this type of research aircraft. Substantial improvement in the stability and control characteristics of the aircraft could, however, be obtained by addition of moderate hub restraint.

All normal helicopter maneuvers were performed during the flight test program within the restricted air space available at the Hughes Culver City plant. All control positions, rates, and attitudes were recorded continuously during flight, and significant quantitative data, corrected to a mid center of gravity and a reference weight, where applicable, were compiled.

FUEL CONSUMPTION

Hover fuel consumption characteristics of the XV-9A are presented in Figure 28. As corrected, both flight and whirl test fuel flows are in excellent agreement with predictions at low rotor lift, and tend to be 8 or 10 percent higher than predicted at high lift. This discrepancy is presumed to be the result of component performance irregularities. Since the various detail performance factors are interrelated according to different equations for fuel flow as compared with rotor lift, it is possible for excellent overall agreement to exist on lift while a noticeable discrepancy exists in fuel consumption. Given sufficient testing and analysis, an improved set of component values can be expected to emerge. They should provide simultaneous agreement on all aspects of rotor performance. Table V and Table VI present observed and corrected fuel flow, rotor and gas power for flight test and whirl test, respectively. (For a more comprehensive discussion of these tables, refer to Appendix IV of Reference 4.)

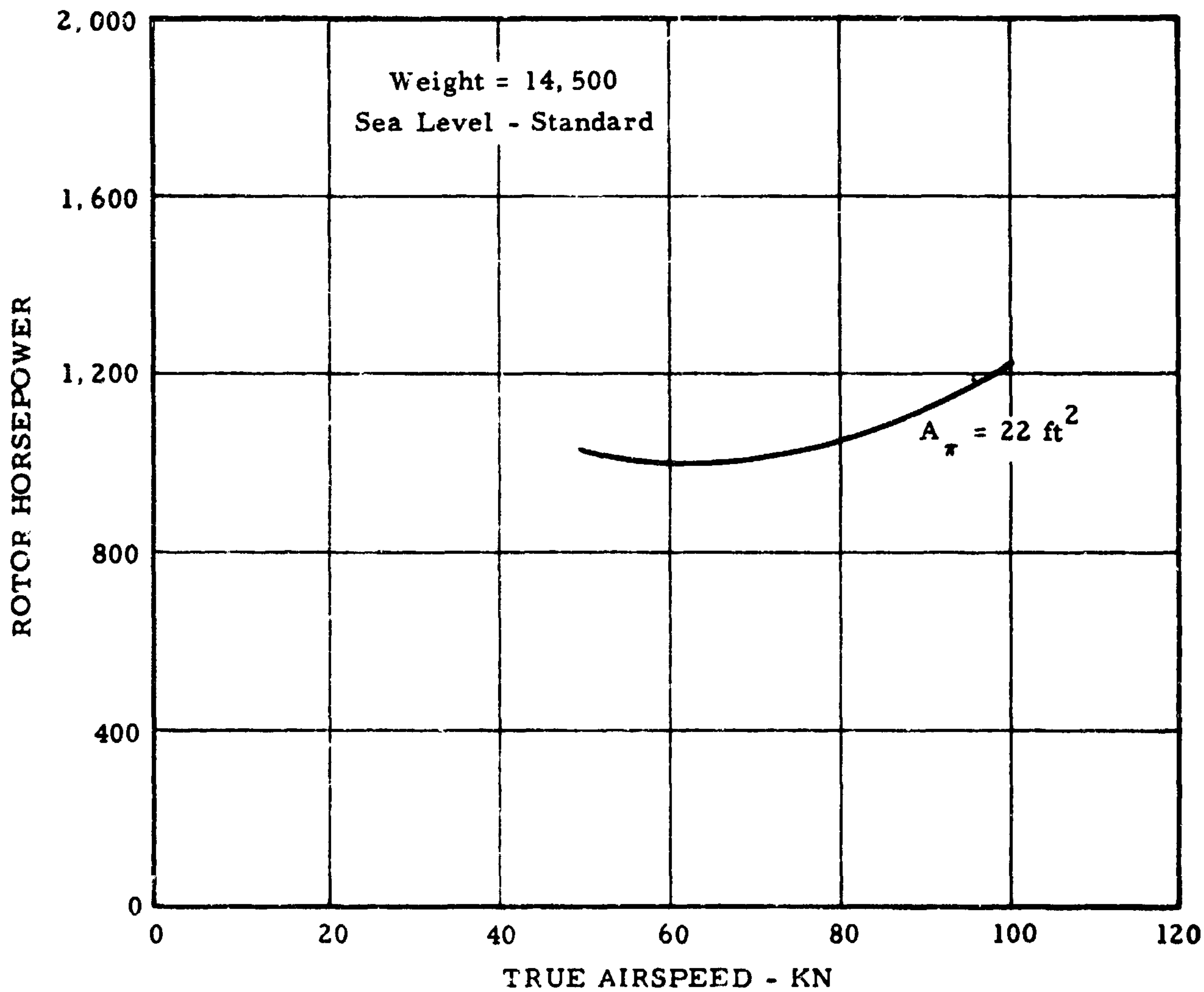


Figure 27. Predicted Power Required With Drag Cleanup.

TABLE V

OBSERVED AND CORRECTED FLIGHT TEST FUEL FLOW AND ROTOR AND GAS POWER

(1) Flight Number	(2) Record Number	(3) TAS (kn)	(4) Hover Gross Weight (lb)	(5) W_2 (lb/sec)	(6) T_5 (deg R)	(7) P_5 (psia)	(8) θ	(9) δ	(10) W_f (lb/hr)	(11) Rotor Tip Pressure Ratio P_7/P_0	(12) RHP	(13) $\frac{RHP}{\delta\sqrt{\theta}}$	(14) SFC (Obs)	(15) $\frac{GHP}{W_A}$	(16) GHP	(17) $\frac{P_5}{\delta}$ (psia)	(18) $\frac{T_5}{\theta}$ (deg R)	(19) $\left(\frac{T_5}{\theta}\right)_{QT}$ (deg R)	(20) $\left(\frac{T_5}{\theta}\right)_{QT} - 519°$ (deg R)	(21) $\left(\frac{Fuel}{Air}\right)_{QT}$	(22) $(W_2)_{QT}$ (lb/sec)	(23) W_f (lb/hr)	(24) SFC (Corr)
10	1,93[illegible]	59.0		42.65	1,270	26.03	0.997	1.0040	1,631	1.725	1,130	1,125	1.445	65.7	2,801	25.70	1,273.5	1,185	666	0.00925	44.22	1,426.3	1.26
	2,002	59.3		43.25	1,256	26.13	0.994	1.0067	1,673	1.710	1,097	1,094	1.525	65.7	2,834	25.00	1,262.8	1,168	649	0.00895	44.97	1,413.6	1.29
	2,094	70.4		43.10	1,273	26.13	0.995	1.0057	1,673	1.719	1,112	1,109	1.504	66.6	2,871	25.95	1,276.0	1,188	665	0.00929	44.67	1,449.1	1.30
	2,153	70.7		44.35	1,264	26.97	0.994	1.0067	1,769	1.823	1,147	1,143	1.542	69.1	3,066	26.60	1,271.9	1,205	686	0.00951	45.57	1,513.4	1.32
	2,236	81.3		43.95	1,299	27.11	0.997	1.0064	1,736	1.933	1,202	1,196	1.444	70.9	3,114	26.25	1,302.4	1,196	677	0.00940	45.87	1,505.5	1.25
	2,295	79.1		43.05	1,256	25.69	0.996	1.0062	1,651	1.734	1,097	1,093	1.505	72.9	2,[illegible]06	25.35	1,260.3	1,175	656	0.00910	44.59	1,417.1	1.29
	2,379	86.4		44.75	1,307	27.61	0.997	1.0060	1,811	1.867	1,322	1,317	1.370	72.8	3,258	27.25	1,311.4	1,221	702	0.00976	46.37	1,580.3	1.20
	2,444	86.4		44.35	1,297	27.46	0.996	1.0064	1,767	1.857	1,212	1,207	1.458	72.1	3,198	27.10	1,301.9	1,218	699	0.00972	45.85	1,556.1	1.28
	2,523	92.8		44.30	1,336	27.61	0.999	1.0064	1,828	1.864	1,370	1,365	1.334	72.6	3,215	27.25	1,340.3	1,222	702	0.00975	46.41	1,581.7	1.15
	2,589	88.7		45.50	1,327	28.54	0.998	1.0070	1,967	1.854	1,382	1,375	1.423	78.6	3,578	28.20	1,333.9	1,245	726	0.01005	47.10	1,652.7	1.20
11	772	00.8	14,800	42.60	1,427	30.31	1.021	1.0072	2,095	1.946	1,532	1,506	1.368	91.4	4,023	29.95	1,397.7	1,290	771	0.01070	45.80	1,711.1	1.12
	1,183	64.1		42.20	1,293	26.28	1.004	0.9835	1,719	1.735	1,170	1,187	1.469	69.9	2,973	26.50	1,287.9	1,203	684	0.00950	43.97	1,458.5	1.25
	1,366	102.4		47.10	1,43[illegible]	3[illegible]	1.005	0.9857	2,365	2.076	1,785	1,807	1.325	100.0	4,865	32.15	1,429.0	1,350	831	0.01159	50.05	2,025.5	1.13
12	1,591	81.4		42.80	1,283	26.28	0.990	0.9755	1,709	1.770	1,1[illegible]	1,216	1.448	70.3	3,008	26.95	1,296.0	1,214	695	0.00965	44.22	1,490.2	1.26
	1,724	79.4		44.20	1,314	27.61	0.990	0.9736	1,833	1.825	1,307	1,350	1.402	77.7	3,435	28.45	1,328.0	1,251	732	0.01018	45.54	1,618.9	1.24
	2,533	100.4		49.60	1,415	31.39	0.995	0.9756	2,323	2.039	1,830	1,851	1.269	97.7	4,749	28.60	1,421.5	1,255	736	0.01020	51.72	1,842.3	1.01
	2,618	92.5		46.40	1,357	29.18	0.992	0.9734	2,056	1.904	1,502	1,548	1.369	86.3	4,004	30.00	1,360.5	1,290	771	0.01070	47.79	1,785.4	1.19
	2,692	55.4		41.90	1,254	24.76	0.988	0.9696	1,627	1.696	1,051	1,090	1.548	63.4	2,658	25.55	1,269.0	1,180	661	0.00920	43.45	1,396.0	1.33
	1,114	00.0	14,625	44.10	1,406	30.16	1.012	1.0063	2,040	1.977	1,560	1,541	1.308	89.1	3,93[illegible]	30.00	1,388.5	1,290	771	0.01070	45.75	1,709.2	1.10
14	724	91.9		42.90	1,374	28.78	1.013	1.0063	1,923	1.910	1,420	1,403	1.354	82.9	3,555	28.60	1,356.5	1,255	739	0.01020	44.60	1,588.7	1.12
	785	90.[illegible]		43.00	1,355	28.61	1.009	1.0050	1,916	1.831	1,342	1,329	1.428	80.6	3,465	28.55	1,342.0	1,253	734	0.01000	44.50	1,553.9	1.16
	876	101.4		43.30	1,384	29.62	1.013	1.0047	2,063	1.968	1,502	1,486	1.360	85.7	3,711	29.50	1,371.0	1,278	759	0.01050	44.85	1,646.0	1.10
	910	10[illegible].3		45.10	1,[illegible]87	30.04	1.009	1.0056	2,115	1.961	1,571	1,554	1.346	87.4	3,943	29.90	1,374.5	1,289	770	0.01070	46.5[illegible]	1,739.9	1.11
15	1,4[illegible]8	[illegible]0.0	14,675	43.[illegible]	1,452	3[illegible].16	1.051	1.0058	2,115	1.935	1,603	1,555	1.319	92.0	4,011	30.00	1,380.0	1,290	771	0.01070	45.10	1,684.9	1.05
	1,[illegible]16	00.0	14,608	42.40	1,456	29.[illegible]4	1.055	1.0056	2,098	1.906	1,462	1,416	1.435	90.9	3,852	29.55	1,373.5	1,27[illegible]	760	0.01050	43.9[illegible]	1,614.4	1.10
[illegible]	444	00.0	14,986	44.00	1,423	30.7[illegible]	1.018	1.0048	2,090	1.965	1,507	1,487	1.367	93.[illegible]	4,098	30.63	1,398.0	1,308	789	0.01098	45.49	1,744.1	1.16
	724	30.0	14,892	42.90	1,410	30.10	1.018	1.0053	1,989	1.888	1,411	1,392	1.410	8[illegible]	3,849	29.95	1,386.0	1,290	771	0.01070	44.47	1,661.4	1.18
	1,447	30	14,267	43.50	1,410	30.00	1.[illegible]24	1.0048	2,056	1.910	1,438	1,456	1.362	88.6	3,453	29.85	1,375.0	1,287	768	0.01062	44.96	1,667.6	1.12
	2,031	[illegible].0	13,066	42.30	1,381	28.9[illegible]	1.127	1.0041	1,869	1.885	1,385	1,362	1.364	83.4	3,529	28.75	1,345.0	1,258	739	0.01023	43.74	1,562.4	1.13
	[illegible]	[illegible]0.0	12,902	41.20	1,372	28.40	1.026	1.0043	1,933	1.827	1,300	1,278	1.395	81.1	3,343	28.30	1,337.0	1,248	72[illegible]	0.01011	42.64	1,505.2	1.16
	[illegible]592	[illegible].0	12,642	[illegible].83	1,303	29.10	1.02[illegible]	1.0345	[illegible]934	1.889	1,417	1,395	1.365	84.0	3,595	29.00	1,348.0	1,263	746	0.01035	44.18	1,598.4	1.13
	2,[illegible]74	[illegible].5		39.90	1,297	26.10	1.003	0.9814	1,658	1.727	1,061	1,079	1.544	69.7	2,782	26.55	1,293.0	1,204	685	0.00950	41.34	1,371.2	1.29
	2,316	8[illegible]		40.7[illegible]	1,326	27.30	1.003	0.9799	1,723	1.801	1,194	1,216	1.443	76.6	3,116	27.90	1,322.0	1,237	718	0.00999	42.07	1,467.8	1.23
	2,369	92.2		42.40	1,362	29.70	1.001	0.9772	1,872	1.697	1,383	1,411	1.354	84.6	3,586	29.30	1,361.0	1,273	754	0.01048	43.63	1,604.2	1.16
	2,425	9[illegible].5		43.10	1,387	29.30	1.050	0.9765	1,956	1.91[illegible]	1,458	1,494	1.342	88.0	3,793	29.95	1,387.0	1,290	771	0.01070	44.69	1,669.6	1.1[illegible]
	2,4[illegible]	106.2		45.80	1,443	31.50	1.000	0.9758	2,210	2.114	1,798	1,892	1.240	100.6	4,606	32.29	1,442.0	1,352	833	0.01157	47.31	1,914.6	1.06
18	2,071	[illegible].0		43.70	1,363	29.96	1.003	0.9926	2,325	1.963	1,510	1,518	1.341	88.6	3,671	30.35	1,379.0	1,292	773	0.01075	45.15	1,691.8	1.12
	2,1[illegible]6	106.0		44.70	1,498	31.00	1.003	0.9912	2,130	2.033	1,642	1,654	1.297	94.3	4,215	31.30	1,404.0	1,325	806	0.01120	46.01	1,799.5	1.10
[illegible]	4,[illegible]66	00.0	14,720	44.00	1,402	30.3[illegible]	1.034	1.0145	2,069	1.989	1,567	1,520	1.320	88.6	3,897	29.92	1,356.0	1,289	770	0.01070	45.12	1,685.7	1.08
	4,661	00.0	14,280	43.20	1,406	30.00	1.043	1.0140	2,059	1.946	1,501	1,449	1.372	86.9	3,752	29.53	1,347.0	1,279	760	0.01052	44.34	1,629.1	1.09
2[illegible]	[illegible]60	90.0	14,[illegible]70	43.10	1,427	29.90	1.045	1.0114	2,064	1.928	1,502	1,454	1.374	88.6	3,817	29.85	1,368.0	1,287	766	0.01057	44.41	1,655.5	1.10
	[illegible]178	102.5		44.60	1,444	31.00	1.026	0.9912	2,168	2.007	1,651	1,647	1.311	96.6	4,307	31.05	1,408.0	1,316	79[illegible]	0.011[illegible]5	46.09	1,794.7	1.09

(1) Flight number

(2) Test point

(3) True airspeed - kn

(4) Hover gross weight for indicated hover points

(5) Engine mass flow - total - lb/sec

(6) Engine discharge temperature - deg R

(7) Engine discharge pressure - psia

(8) Engine inlet temperature ratio

(9) Atmospheric pressure ratio

(10) Total fuel flow - lb/hr

(11) Rotor tip pressure ratio

(12) Rotor horsepower RHP $= \frac{0.97\, W_2 (C_{V_V} V_j - V_T) V_T}{g \times 550}$

where

0.97 is factor allowing 3 percent leakage through diverter valves, etc.

V_j = f blade tip pressure (11) and temperature ((6) - 10 deg)

C_{V_V} = velocity coefficient

(13) Referred rotor horsepower

(14) Specific fuel consumption - lb/hr/hp (10)/(12)

(15) Gas horsepower/lb/sec air W_2 - function of P_5, (7), (6), and (8)

(16) Gas horsepower = airflow × GHP/W_2 × 1. GHP = W_2 × GHP/W_2

(17) (7)/(9)

(18) (6)/(8)

(19) $(T_5/\theta)_{QT}$ = referred temperature for QT engines. Read from Figure 22 at given P_5/δ (17)

(20) As indicated

(21) Taken at (20) from Figure 26 for QT rather than YT engine

(22) QT airflow is determined by increasing YT engine airflow (5) by inverse ratio of square root of T_5, that is, $\sqrt{\frac{(T_5/\theta)}{(T_5/\theta)_{QT}}}$ or $\sqrt{\frac{(18)}{(19)}}$

(23) The fuel flow is the product of fuel/air ratio × airflow × leakage factor, or (21) × (22) × 0.97 × 3600 sec/hr

(24) Specific fuel consumption - lb/hr/hp (23)/(13)

TABLE VI
OBSERVED AND CORRECTED WHIRL TEST FUEL FLOW

① Run	② Point	③ Lift (lb)	④ W_2 (lb/sec)	⑤ $\frac{P_5}{\delta}$ (psia)	⑥ $\frac{T_5}{\theta}$ (deg R)	⑦ $\frac{T_5}{\theta} - 519°$ (deg R)	⑧ $\left(\frac{Fuel}{Air}\right)$ Figure 26	⑨ W_f (lb/sec)	⑩ $\left(\frac{T_5}{\theta}\right)_{QT}$ (deg R)	⑪ $\left(\frac{T_5}{\theta}\right)_{QT} - 519°$ (deg R)	⑫ $\left(\frac{Fuel}{Air}\right)_{QT}$	⑬ $(W_2)_{QT}$ (lb/sec)	⑭ $0.97\ W_f$ (Corrected for leaks)
22	8	16,000	43.50	30.9	1,453	934	0.01300	2,036	1,315	796	0.01110	45.7	1,770
	9	18,000	46.15	33.4	1,505	986	0.01380	2,293	1,382	863	0.01210	48.2	2,035
	10	18,500	46.50	33.7	1,535	1,016	0.01420	2,377	1,392	873	0.01220	48.8	2,080
	11	19,000	46.50	35.2	1,535	1,016	0.01420	2,377	1,433	914	0.01280	48.1	2,150
23	7	17,500	45.80	33.0	1,500	981	0.01370	2,259	1,372	853	0.01200	47.8	2,005
	8	19,200	47.35	34.8	1,561	1,042	0.01460	2,489	1,422	903	0.01270	49.6	2,200
	9	19,000	46.95	34.9	1,551	1,032	0.01440	2,434	1,425	906	0.01270	49.0	2,170
	10	19,000	47.00	34.8	1,551	1,032	0.01450	2,453	1,422	903	0.01270	49.0	2,170
	11	19,300	46.80	35.0	1,561	1,042	0.01460	2,460	1,427	908	0.01270	49.0	2,170
24	9	15,500	42.2[illegible]	29.9	1,422	903	0.01260	1,915	•1,288	769	0.01070	44.4	1,660
	9a	19,500	41.85	34.6	1,555	1,036	0.01450	2,446	1,416	897	0.01250	49.1	2,140
26	10a	4,700	33.80	22.4	1,198	679	0.00940	1,144	1,115	596	0.00826	35.1	1,010
	11a	11,600	38.75	26.5	1,323	809	0.01150	1,604	1,202	683	0.00951	40.7	1,350
	b	14,000	40.90	28.3	1,360	841	0.01170	1,723	1,246	727	0.01010	42.7	1,505
	c	16,000	44.20	31.[illegible]	1,426	907	0.01260	2,005	1,317	798	0.01110	46.0	1,785
28	8a	12,000	38.76	26.7	1,325	806	0.01115	1,556	1,207	688	0.00956	40.6	1,355
	b	14,600	41.50	28.6	1,381	862	0.01200	1,793	1,255	736	0.01025	43.6	1,560
	c	15,800	43.82	30.6	1,429	910	0.01270	2,003	1,306	787	0.01096	45.8	1,755
	d	20,000	48.59	35.0	1,540	1,021	0.01430	2,501	1,427	908	0.01270	50.5	2,235
	e	20,800	49.45	36.0	1,570	1,051	0.01470	2,617	1,455	936	0.01311	51.4	2,355
30	14a	15,200	43.62	30.2	1,402	883	0.01230	1,931	1,296	771	0.01075	45.4	1,705
	b	19,250	47.33	34.1	1,505	986	0.01380	2,351	1,402	883	0.01235	49.0	2,110
	c	20,600	49.58	36.4	1,568	1,049	0.04700	2,624	1,467	948	0.01380	51.3	2,380
31	6a	3,700	33.97	22.5	1,210	691	0.00960	1,171	1,117	598	0.00830	35.3	1,024
	b	6,500	33.95	22.9	1,220	701	0.00980	1,198	1,125	606	0.00890	35.4	1,040
	c	7,500	36.73	24.4	1,250	731	0.01010	1,336	1,155	636	0.00880	38.2	1,180
	d	9,700	37.41	25.2	1,282	763	0.01050	1,414	1,172	653	0.00910	39.1	1,240
	e	12,300	40.12	27.4	1,335	816	0.01130	1,628	1,224	705	0.00980	41.9	1,435
	f	15,400	44.29	30.8	1,423	904	0.01260	2,009	1,312	793	0.01110	46.1	1,790
	g	15,500	42.53	34.2	1,416	897	0.01250	1,914	1,405	886	0.01240	42.7	1,850
	h	18,500	47.22	34.3	1,510	991	0.01390	2,363	1,407	888	0.01240	48.9	2,120
	i	13,000	45.20	32.9	1,480	961	0.01340	2,180	1,369	850	0.01190	46.9	1,950
	k	20,300	49.39	36.5	1,561	1,042	0.01460	2,596	1,470	951	0.01340	50.9	2,380
	16a	19,500	46.48	34.0	1,505	986	0.01375	2,301	1,400	881	0.01230	48.2	2,070
	b	23,000	50.33	38.3	1,614	1,095	0.01535	2,792	1,520	1,001	0.01405	51.6	2,530
	c	23,000	50.46	38.7	1,630	1,111	0.01560	2,834	1,532	1,013	0.01428	52.0	2,590

① Run number
② Test point
③ Lift
④ Engine mass flow, total, lb/sec
⑤ Referred engine pressure
⑥ Referred engine temperature
⑦ As indicated
⑧ Fuel/air (Figure 26) at ⑦
⑨ Fuel flow $W_2 \times F/A \times 3,600$, or ④ × ⑧ × 3,600
⑩ Referred temperature for QT engine. Read from Figure 22 at given P_5/δ, ⑤
⑪ As indicated
⑫ Fuel/air (Figure 26) at ⑪ for QT engine
⑬ QT airflow is determined by increasing YT engine airflow ④ by inverse ratio of square root of T_5, that is, $\sqrt{\frac{(T_5/\theta)}{(T_5/\theta)_{QT}}}$ or $\sqrt{\frac{⑥}{⑩}}$
⑭ Fuel flow $W_2 \times F/A \times 3,600 \times$ leakage factor, or ⑬ × ⑫ × 3,600 × 0.97

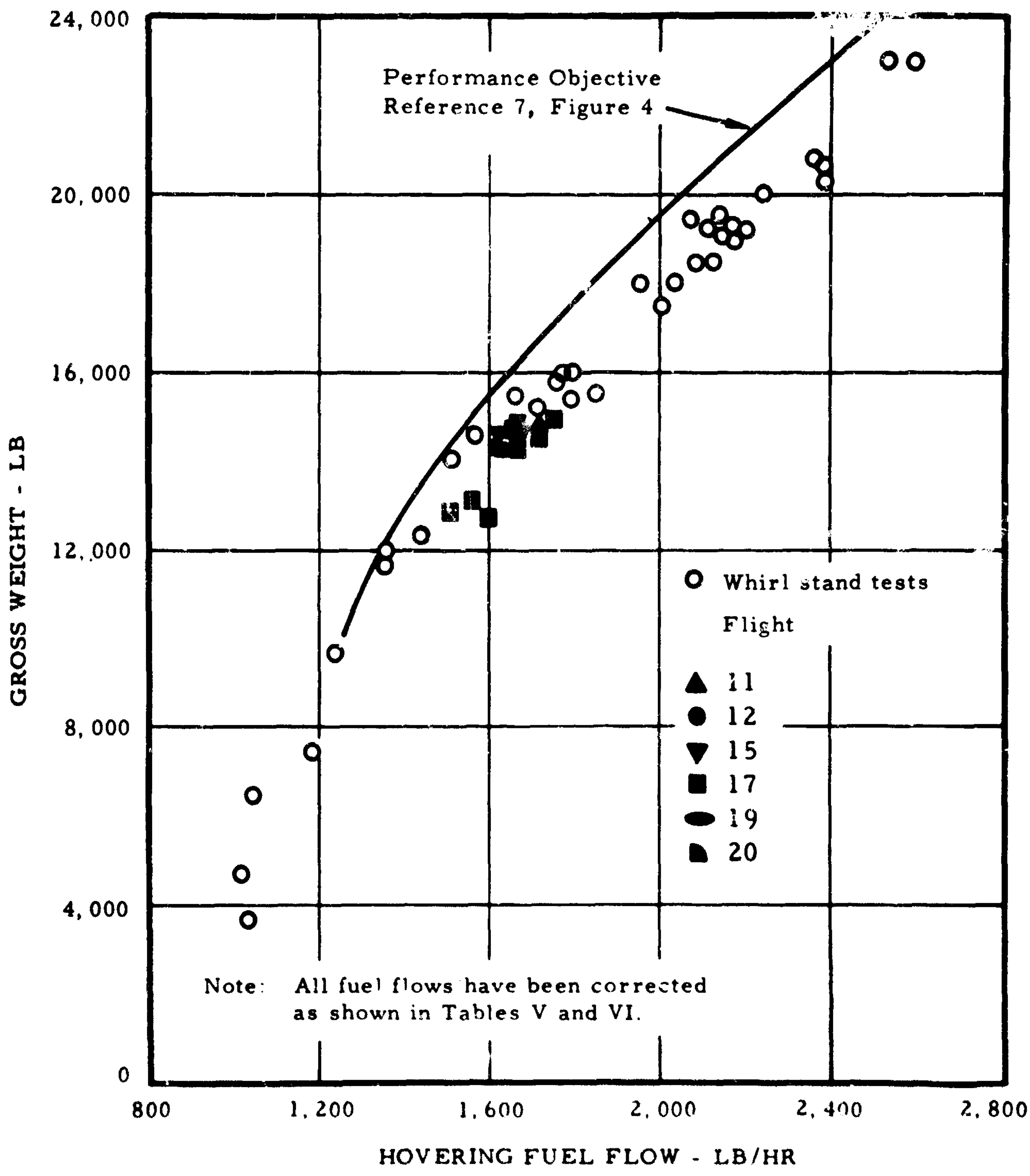


Figure 28. Hovering Fuel Flow Versus Gross Weight.

STRUCTURAL LOADS

Structural loads data were reviewed carefully after each flight to ensure safety of flight. Rotor-blade moments experienced during flight were fairly typical of those experienced on other helicopters. Blade moments were low during hover, fairly high when accelerating or decelerating through the transition region, low at the minimum power speed, and increasing thereafter with speed. In addition to the typical response to aerodynamic excitation, a type of excitation and response was encountered that is typical of chordwise unarticulated blades. The XV-9A rotor blades have a chordwise natural frequency of 1.4 per rev. Motion of the cyclic stick at 0.4 per rev in the nonrotating system will excite 1.4 per rev in the rotating system; 0.4 per rev is 1.6 cps, which is a frequency at which pilots occasionally move the cyclic stick. Only a small content of 0.4 per rev in the stick input waveform is sufficient to excite chordwise bending. The XV-9A pilot inputs at this frequency were reduced by installing cyclic stick centering springs and by reducing servo-valve friction as much as possible, both of which improved the "feel."

Visual inspection of all primary structural components, including the hub, retention straps, and blade spar area at station 91, was accomplished after each flight. Complete blade spar inspection was accomplished at the end of each five flight hours by removal of leading and trailing edge sections. There was no evidence of any structural distress resulting from fatigue or temperature at any time during the flight test program.

STRUCTURAL AND OPERATING TEMPERATURES

Extensive temperature measurements were recorded during all flights, on the rotor components, airframe components, systems, and powerplant. These temperatures were checked after each flight to determine that the various components were within allowable operating temperature range.

Data were read and analyzed, and produced operating temperatures of the following rotor components and systems: (1) blade-tip gas, (2) front and rear spars, (3) flexures, (4) ribs, (5) root-end spar cooling air, (6) outer skins, (7) gas duct walls, (8) rotor shaft, (9) bearings, and (10) articulate duct seals.

Operating temperatures of the following powerplant and airframe components were read and analyzed: (1) engine and engine accessories, (2) engine and diverter-valve bays, (3) lateral pylons and nacelles, (4) rotor bearing housings, (5) aft fuselage and yaw-valve compartment, (6) yaw-duct and Y-duct insulation, (7) yaw-duct and Y-duct bays, and (8) yaw-valve outlet.

None of the temperatures recorded exceeded the design limit temperatures for either the rotor, the powerplant, or the airframe.

VIBRATION

Airframe vibration during flight was predominantly at a frequency of 3 per rev (10 to 12 cps), because of the fuselage resonance near this point. Vibration levels at cockpit were lowest during hover and increased during level flight, with the maximum recorded during transition to forward flight and approach to hover. Average vertical acceleration at the pilot's position was ±0.35 g in level flight and ±0.13 g during hover.

SOUND POWER LEVELS

The overall sound level of the XV-9A during hover was not objectionable to nearby observers. Sound power level measured in the cockpit during flight was 109 decibels. This was not objectionable to the pilot, and radio communications were satisfactory. Sound data were recorded from 50- to 400-foot distance from the aircraft at 100-foot intervals for four azimuth positions. As shown by Figure 29, sound power level measured at 200 feet in hover was 103 decibels.

ROTOR DOWNWASH

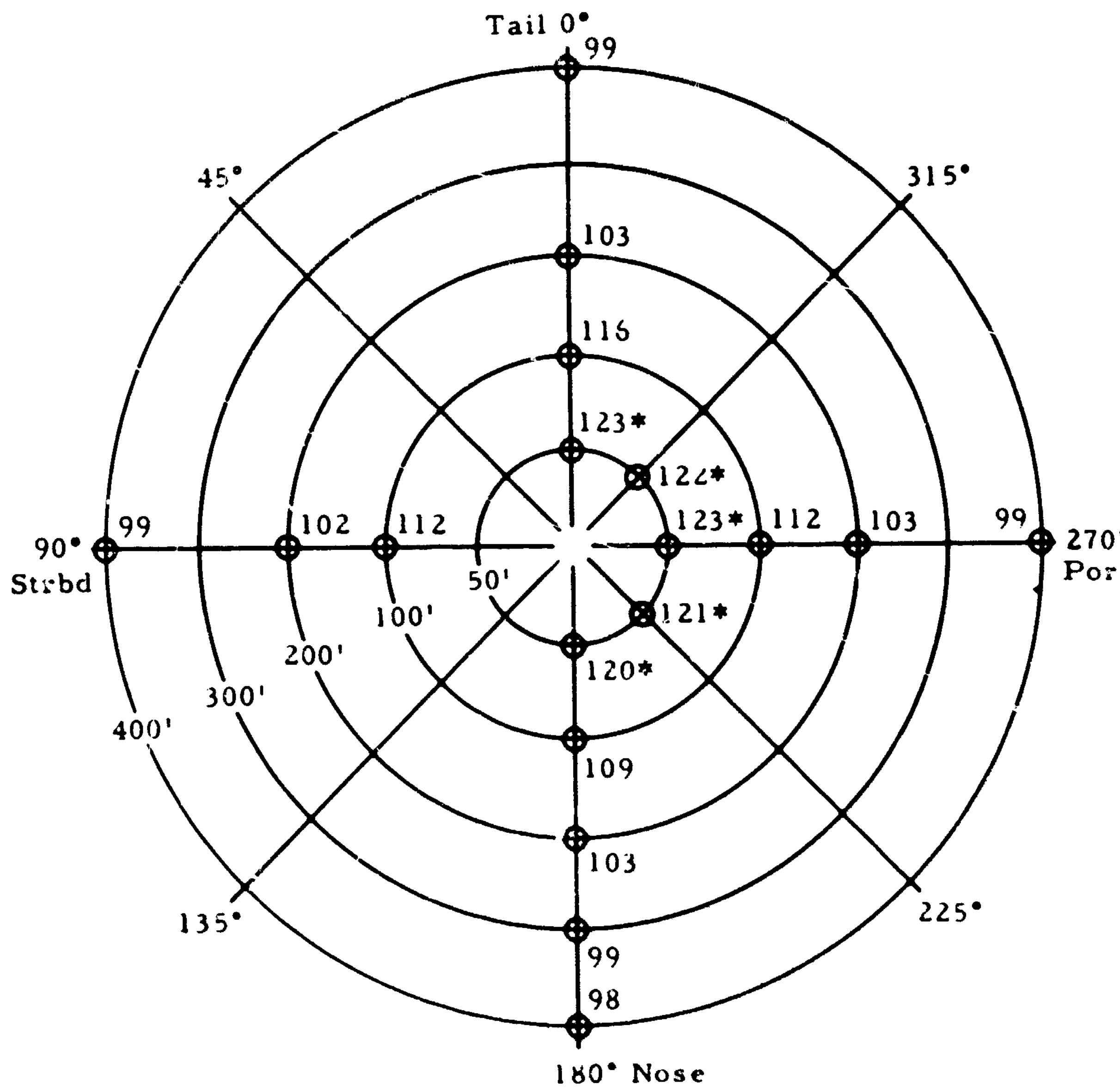
Rotor downwash velocity was measured during hover at approximately 20-foot wheel height. The maximum velocity measured was 73 feet per second at approximately 70-percent blade radius. The aircraft was hovered at 6-foot wheel height over unprepared sod and dirt surfaces that varied from dead to green grass on the sod surface and from loose dirt to small rocks on the dirt surface. There was no engine foreign object damage (FOD) or rotor blade damage incurred. Recirculation effects during hover were of the magnitude of a 5- to 7-degree F rise in engine inlet temperature when hovering in still air.

STATIC ELECTRICITY BUILDUP

The electrostatic voltage buildup and charging current were measured during hover at 20-foot wheel height. The electrostatic charge was low (7 to 9 kilovolts) in comparison with that of shaft-driven helicopters.

Hover at 15-foot altitude, except * at 6 feet

Gross weight = 14,000 ±500 lb



All-pass band shown

Equipment used: General Radio Corporation

1. Octave band analyzer
2. Sound level meter, Type 1551-C, C weighting
3. Graphic level recorder, Type 1521-A
4. Crystal microphone

Figure 29. Sound Pressure Level In Hover.

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13. ABSTRACT

This report summarizes a research program covering the design, fabrication, and test of the XV-9A Hot Cycle Research Aircraft.

Discussion of the program is broken into five major areas: design and fabrication, engine and whirl tests, component testing, ground tests, and flight tests. During the program, conducted from 29 September 1962 through 15 March 1965, the flight feasibility of the Hot Cycle rotor was successfully validated.

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